University of Illinois
at
Urbana-Champaign
Aeronautical and Astronautical Engineering

AAE 241
Aerospace Vehicle Design, Spacecraft Section
Final Project Reports

Volume II
Project Groups 5 through 7

May 1990

P-98

N91-16051

AAE 241 Spacecraft Design Project

Group #5

Spring 1990

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INTRODUCTION

PULSE, Pluto Unmanned Long-Range Scientific Explorer, is an unmanned probe that will do a flyby of Pluto. It is a low weight, relatively low costing vehicle which utilizes mostly off-the-shelf hardware, but not materials or techniques that will be available after 1999.

PULSE will be launched within the first decade of the twenty-first century.

MISSION MANAGEMENT, PLANNING, AND COST

1.1 INTRODUCTION

In the subsystem of mission management, planning, and cost many selections were made. The mission type, trajectory, and launch date were selected. The optimum delta-v and cost of the project were also calculated.

1.2 TYPE OF MISSION

A flyby was the type of mission selected. This selection was made due to its low delta-v, short mission duration, and simplicity, all of which are directly related to this mission's low cost.

Simplicity was a main issue in selecting this mission class. Since there have been no missions to Pluto and Pluto's distance from the Earth is very far, very little is known about Pluto and Charon. Therefore, before a high-cost, elaborate mission can be sent, scientists need more accurate information. A flyby mission is the most efficient way to get the information that is needed.

1.3 TRAJECTORY

The trajectory selected for this mission is a direct Earth to Pluto path. Again, simplicity was an important issue in the selection process. The more complex a mission, the greater the opportunity for something to fail. So by using a direct path, simplicity is optimized.

1.4 MISSION DELTA-V REQUIRED

The delta-v required for the PULSE mission is 8.606 kilometers per second from a parking orbit around Earth.

1.5 MISSION TIMELINE

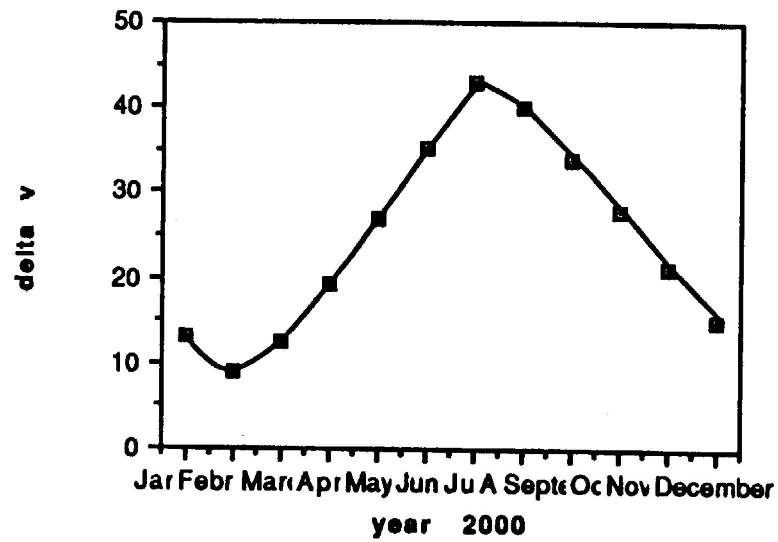
The launch date was determined to be January 30, 2003. The arrival at Pluto was determined to be February 1, 2019. The mission length is 16.005 years. The launch date was chosen by selecting the date with the optimum delta-v. To obtain a selection of dates, data was input for the first of every month of every year from the year 2000 to the year 2010. (Graph 1.1)

1.6 COSTING

The costing process of this mission was done in several steps. First, for each subsystem, the direct labor hours and the recurring labor hours were calculated. This was done by several different formulas that used the mass of each subsystem and the number of spacecraft. The number of spacecraft costed were four, three of which are flight ready and one which is used in an integrated ground test system.

Next, for each subsystem, an inheritance class had to be defined. Class One is an off-the-shelf buy. Class Two is an exact repeat of a subsystem. A Class Three inheritance is the use of a previous subsystem with minor modifications. A Class Four inheritance is also a use of a previous subsystem, but with

Graph 1.1 Illustrates ΔV for 1st of every month in the year 2000.



major modifications. Finally, a Class Five inheritance is an entirely new subsystem. (Table 1.1)

The next step was to convert labor hours into labor cost. Then the labor costs were converted into total costs. The conversion factors were given in Fiscal Year 1977 which needed to be converted to Fiscal Year 1988. This was done by using a consumer price index. The consumer price index for all items in 1977, with a base of 1967=100, was 181.5. The consumer price index for all items in 1988, with a base of 1967=100, was 354.3. (Appendix 1).

Finally, these conversions were made for each subsystem and then added to obtain the total cost of the project. (Table 1.2). The total cost of the PULSE project is about 1.7 billion dollars.

1.7 EFFECTS ON SUBSYSTEMS

Many of the selections made affected the selections of the other subsystems. The selecting of a flyby affected the science instrument selection. Because the mission is a flyby, only instruments which can be used quickly and at a distance could be used. The power and propulsion subsystem was also affected. By utilizing a flyby instead of an orbiter or a lander, less fuel was needed. These factors also affect the design of the structure.

The length of the mission and the trajectory selected also affected the other subsystems. Due to the length of the mission, 16.005 years, science instruments and other materials which lifetimes exceed 16.005 years had to be selected. These

Table 1.1

SUBSYSTEM INHERITANCE CLASS

Category	Inheritance
Structure	5
Thermal Control	1
Propulsion	1
Attitude & Articulation	3
Telecommunications	2
Antennas	2
Command & Data Handling	1
RTG Power	2
Line-Scan Imaging	2
Particle & Field Instruments	1
Remote Sensing Instruments	1

Table 1.2

Costing for PULSE

Category	Cost (FY 88 Dollars)
Structure	59,988,162.98
Thermal Control	11,037,938.33
Propulsion	412,927,670.50
Attitude & Articulation	62,614,609.37
Telecommunications	64,098,191.33
Antennas	13,043,018.66
Command & Data Handling	24,500,108.53
RTG Power	37,396,446.55
Line-Scan Imaging	170,454,335.10
Particle & Field Instruments	71,222,537.72
Remote Sensing Instruments	29,154,302.64
System Support & Ground Equipment	280,062,535.20
Launch + 30 Days Ops & Ground S/W	57,185,698.78
Image Data Development	6,957,007.47
Science Data Development	11,487,733.40
Program Management	17,365,267.83
Flight Operations	258,722,216.60
Data Analysis	115,984,760.70
TOTAL	1,704,192,542.00

selections affect the amount of fuel needed and the design of the structure.

1.8 CONCLUSION

Within the mission management, planning, and cost subsystem, many important selections were made. The PULSE mission is a flyby with a mission duration of 16.005 years. The launch date is January 30, 2003. PULSE is scheduled to arrive at Pluto on February 1, 2019. This mission requires an 8.606 delta-v from a parking orbit.

APPENDIX 1

Fiscal Year '77 to Fiscal Year '88 Conversion:

(Total Cost)(FY88 dollars)/FY77 dollars = Total Cost for the Fiscal Year 1988

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PULSE ATTITUDE AND CONTROL SYSTEMS (AACS)

1. INTRODUCTION

Pulse is a three-axis stabilized spacecraft utilizing solid state sensors and reaction jets to provide control moments. The control hardware utilizes advances in microprocessor accuracy capability, reliability and efficiency.

2. AACS FUNCTIONS

For the purposes of identifying AACS requirements, three main mission phases are distinguished. These phases and their associated AACS tasks are listed below.

GEOSTATIONARY EARTH ORBIT (GEO)

The launch vehicle and upper stage will insert PULSE into GEO. During this phase the deployment, of the booms, the spacecraft attitude, and it's insertion into it's inter planetary trajectory will all be controlled from the ground via the low gain antenna.

CRUISE PHASE

During the cruise phase of the mission, both the determination and control of the spacecraft attitude will be autonomous. The main spacecraft control requirement is that of maintaining the antenna pointing within one degree of earth as the spacecraft progresses along it's trajectory. This task can be viewed as a continuous maneuver of low angular rate or as stabilization of the spacecraft in a non-inertial reference frame.

ENCOUNTER PHASE

The accuracy required of the AACS is much greater as it now must control the scanning of the scientific instruments. The antenna pointing requirement must be maintained both during and after the encounter while stored data from the science instruments is transmitted to earth.

3. DESIGN OF AACS

The primary movers in design of attitude determination and control systems are reliability and low cost. The emphasis of current research in spacecraft attitude determination and control is in the area of control systems, where much of the fundamental work remains incomplete (Ref. p 714-715). Therefore, in the area of attitude determination, use of off the shelf components that have been flight tested on interplanetary missions of long duration, is maximized. Some of the components, such as rate integrating gyros and servomotors will be directly implemented. In other cases, such as that of optical sensors, hardware that is already under development will be utilized. This use of developing technology is justified where it makes use of advance in solid state technology to improve performance yet can still be integrated into flight tested attitude determination systems

(Ref. 2). The rapid advances in microprocessor technology that have taken place since the design of the last interplanetary probes will also be made use of. Modern microprocessors once space hardened, will permit the implementation of control laws which greatly improve performance parameters of the AACS (Ref. 3). The computing power and memory capability available will permit utilization of artificial intelligence (AI) applications such as expert systems. While their low processing power precludes their use in low level control loops they will be useful in the areas of system checkouts and trouble shooting (Ref. 4). Previous missions have employed a fault recovery ability which monitors the system and placed the spacecraft in a safe mode in the event of failure. However, ground control was necessary to reconfigure and reprogram the system before the mission could resume. An expert system would be able to not only diagnose the fault, but to make and implement decisions to rectify the failure.

ATTITUDE DETERMINATION

Figure 1 is an overview of sensor types (Ref. 4). The relevant criteria are that the sensors chosen must be applicable to three-axis stabilized spacecraft in eccentric orbits and have at least medium accuracy. The sensors to be utilized on PULSE are the Yaw Sun Sensor (YSS) and the Solid State Detector (SSD) star tracker.

The Yaw Sun Sensor under development utilizes a charge coupled device (CCD) detector. This sensor is easily integrated

PAPER REFERENCE	SENSOR	TARGETS: EARTH	SUN	STARS	PLANETS	ORBITS: LOW ORBIT	TRANSFER	GEOSTATIONARY	ECCENTRIC	MODE: SPINNING S/C	3.AXIS STABILIZ	ACCURACY: LOW	MEDIUM	HIGH	NOTES
21	HORIZON CROSSING INDICATOR	*				*	*			*		*			FLOWN
22	DUAL BEAM IR EARTH SENSOR	*					*	*		*			*		FLOWN
23	IR FAN BEAM SENSOR	*	*		*		*		*	*			*		QUALIFIED
24	IR PITCH / ROLL SENSOR (STATIC)	*						*			*		*	*	FLOWN
25	IR PITCH / ROLL SENSOR (SCANNING)	*						*			*		*	*	FLOWN
26	LOW ORBIT IR EARTH SENSOR	*				*					*	*	*		DEVELOPMENT
27	YAW EARTH SENSOR	*				*					*		1) *		DEVELOPMENT
28	ALBEDO SENSOR	*	*		*		*		*	*			*	*	QUALIFIED
31	FAN BEAM SENSOR		*			*	*	*	*	*			*		FLOWN
32	HIGH ACCURACY SUN SENSOR		*			*		*	*		*			*	DEVELOPMENT
33	YAW SUN SENSOR		*			*		*	*		*		1) *		STUDY
41	STAR MAPPER			*		*		*	*	*			*	*	QUALIFIED
42	STAR TRACKER (IDT)		*	*	*	*		*	*		*			*	QUALIFIED
43	STAR TRACKER (SOLID STATE DET.)			*	*	*		*	*		*		*	*	DEVELOPMENT
44	RATE SENSOR			*		*		*	*	*			*		STUDY

1) TARGET TO BE VERIFIED

Fig. 1 OVERVIEW OF SENSORS PROPERTIES

REF 5

into optico-inertial systems. In addition sensors being developed on this baseline can be radiation hardened. and can utilize hybrid electronics to minimize weight and reduce dimensions. Finally it may be employed as a high sensitivity sun sensor to aim at sources of light much fainter than the sun (Ref. 6). In this capacity as a planet sensor it may be used to generate error signals to drive the servomechanism which controls the instrument scanning platform.

The Sun Sensor provides only the orientation of a sun pointing vector to the spacecraft. A star tracker which tracks the star Canopus, near the south ecliptic pole provides additional input which uniquely fixes the spacecraft attitude. Such sun-canopus systems have been flown on the mariner, surveyor and lunar orbiter missions (Ref. 1 pp.189). The CCD star tracker to be used features inherent geometric stability, low voltage operation and high reliability (Ref. 5). Because the angular displacements between the earth, sun and canopus are small and the high gain antenna must be earth pointed. The optical sensors must be place on the antenna rim to avoid blocking their field of view.

Rate integrating gyros can be used off the shelf and be integrated with the optical sensors into an optico-inertial attitude measurement system. The gyros will be placed on the body of the spacecraft and on the scan platform to measure pointing of the science instruments.

The gyros will be used for short term attitude measurement and the optical sensors will be used for long term measurement and calibration of the gyros.

CONTROL HARDWARE

A high precision microprocessor implemented control system accepts the angular displacement, rate and disturbing torque from the sensors above. The control law produces time optimal recovery from large angle errors and can obtain stable control with disturbing accelerations approaching the control torque. The control law also incorporates fuel optimal slewing through unlimited angles. Steady state limit cycles in the arc-second region are attainable for precise control during the encounter phase (Ref. 3).

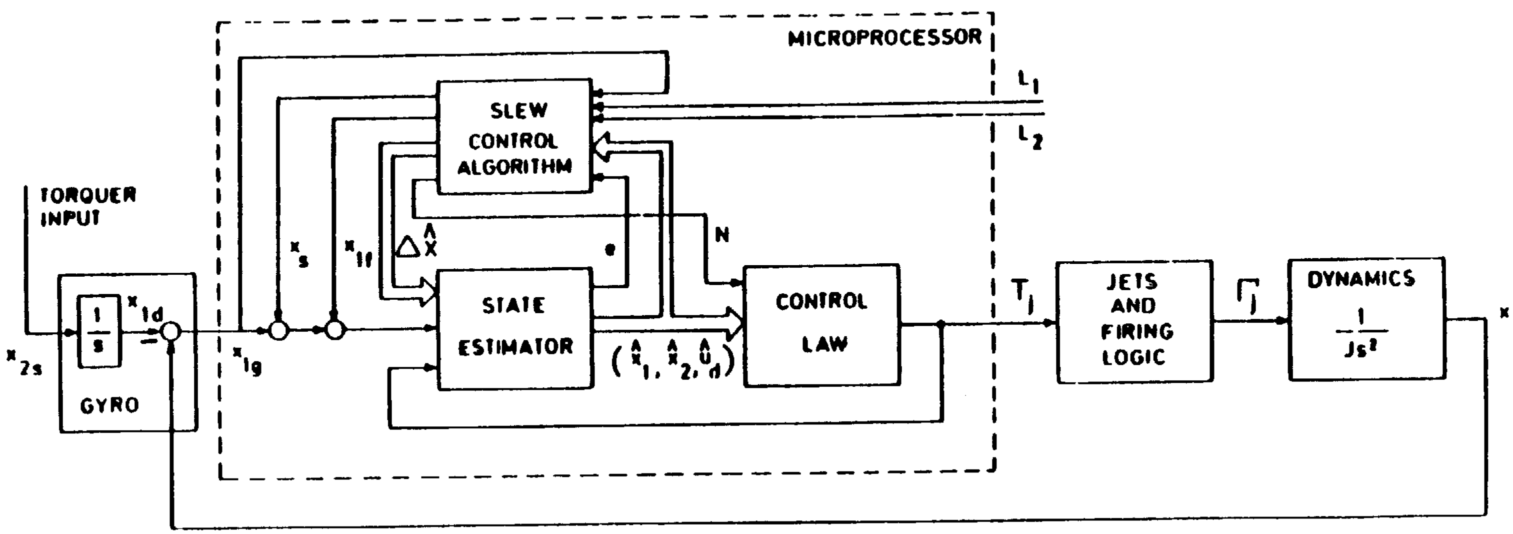
Fig 2 shows a block diagram of the control loop. The state estimator generates a state vector consisting of angular rate, displacement and disturbance torque. The slew algorithm optimizes fuel consumption. The control law controls timing of jet firing.

For the PULSE mission it is required that the microprocessor also generate the command input. This requires on board calculation of the proper earth pointing angle at all stages of the mission. Another difficulty may arise in controlling. The scanning of the science from integrated gyro and accelerometer data. A separate planet sensor on the scan platform ma be required to provide an error signal to the servomotor which controls the platform.

Torquer Selection

There are two types of torquers available for a field free

Attitude control system for microprocessor implementation



(Fig. 2) Control loop

REF 3

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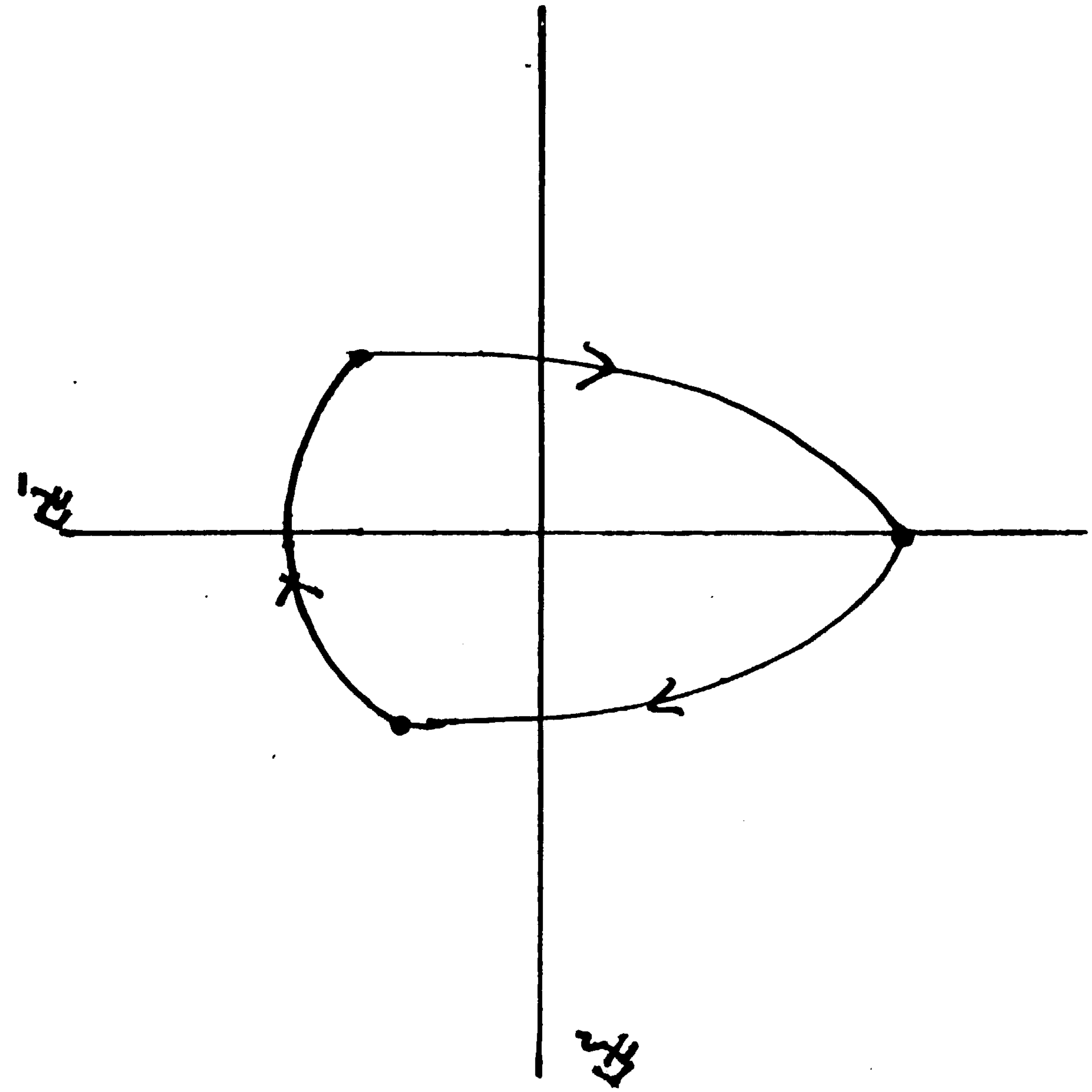

(Fig. 3) Ideal limit cycle

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environment: momentum exchange and mass expulsion. Gas jets are the only viable alternative for missions of this duration (Ref.7) estimates of spacecraft moment of inertia and an assumed impulse bit of .005 s and a limit cycle deadband of 1 degree were used to estimate total impulse required for maintaining antenna pointing during cruise. This assumes that any maneuvering requirements are negligible compared to the essentially continuous limit cycle (Ref. 8)(Appendix A). The total impulse led to a trade study among possible propellants. Cold gas, hydrazine and bipropellants were the candidates. Bipropellants and augmented hydrazine were eliminated because of the required complexity. Fig 4 shows a trade analysis for the propellants. This shows the optimum propellant is hydrazine.

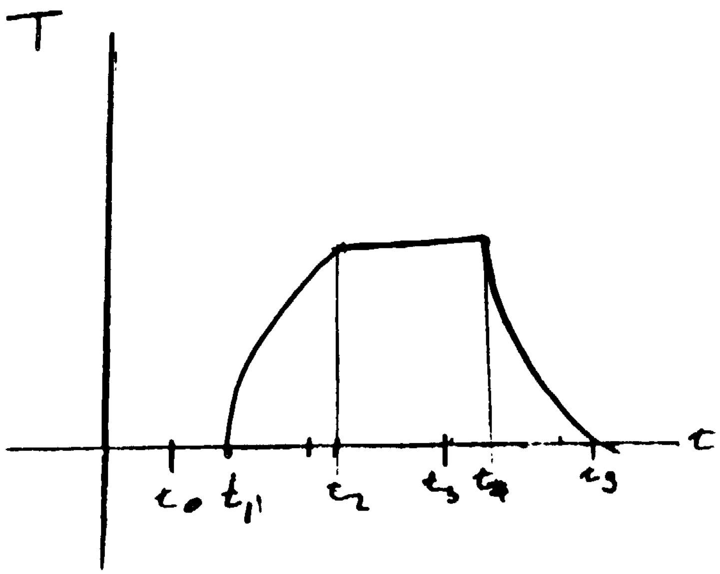
This analysis assumes a torque free environment. To check the validity of this assumption an estimate of the maximum solar torque was made. This torque was shown to be negligible when compared to the control torque thus justifying the assumption (Appendix B).

Other possible errors are introduced into the analysis by changes in thruster performance over time, propellant sloshing in the tank, and inaccurate modeling of thrust profile.

	cold gas	hydrazine	Aug. Hydrazine	Bi propellant
Isp	~60	~100	~200	~300
Minimum Imp. bit	~.0001	~.001	~.01	.015
Corrosive Exhaust	~ no	yes	no	no
Added Complexity	no	no	yes	yes
Mp				

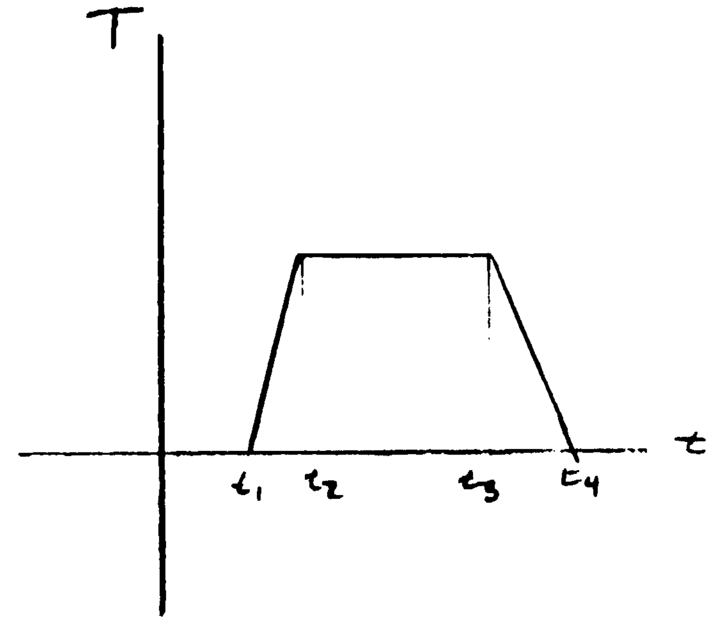
using L=2m Theta= 1 degree I=2000 N*s T = 5*E3

Figure 4



Actual Thrust Profile

(Fig. 5)

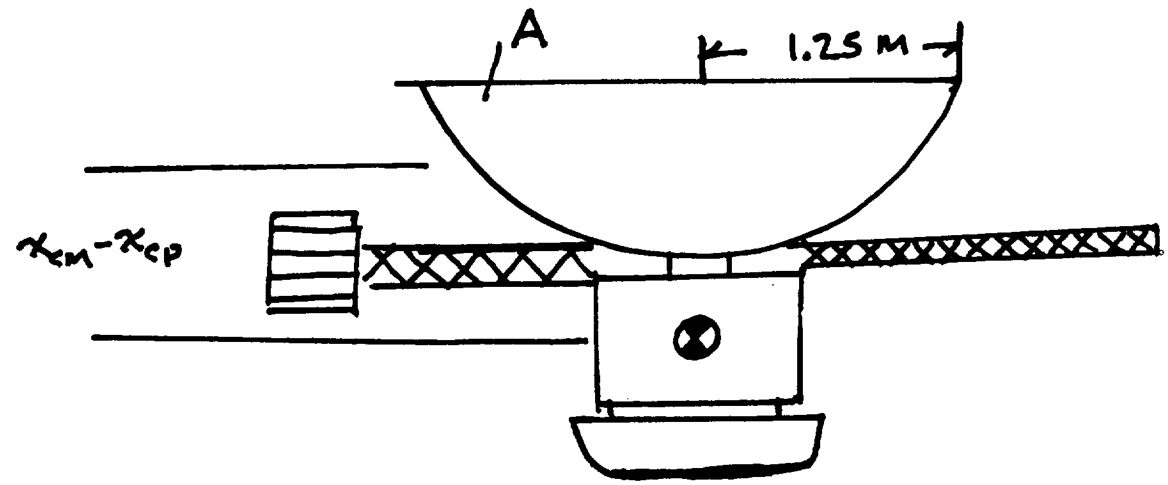


Trapezoidal approximation for thrust profile

REF 1.

APPENDIX B

$$T_{SR} = C_A P_s A (x_{cm} - x_{cp})/2$$



$x_{cm} - x_{cp} = .4\,M$

$A \cong \dfrac{\pi (1.25M)^2}{3} \cong 1.6\,M^2$

$P_s = 4.62 \times 10^{-9}\ \dfrac{N}{M^2}$

$C_A = .7$

$$T_{SR} \cong 10^{-8}\ N \cdot M$$

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3.0 Science

3.1 Mission Objectives

The primary objective for this unmanned, scientific study of Plutonian space is to expand upon our current knowledge of the Pluto-Charon system. This will be accomplished by obtaining and returning information concerning our three scientific objectives which are listed and prioritized in Table 3.1. Each of these objectives will be investigated through the use of the PULSE Experimental Package and the radio science equipment aboard the probe.

Table 3.1
Scientific Objectives of the PULSE probe
1. Investigate Plutonian Characteristics
2. Investigate Satellite Characteristics
3. Investigate Planetary and Interplanetary Particles and fields.

The investigation of each of these scientific objectives is the major concern of this mission. Since no probe has visited Plutonian Space, little is known about the planet Pluto or its satellite Charon. However the scientific community has conducted recent studies concerning the Pluto-Charon system. The knowledge gained from these studies was one of the determining factors for instrument selection aboard the PULSE probe. Although these studies have given us some new information, none of the information can be considered conclusive until a closer investigation is conducted.

3.2 Science Objectives

3.2.1 Plutonian Characteristics

One characteristic of Pluto which must be investigated is the atmosphere. Astronomers have found that Pluto does have a dilute atmosphere which extends several hundred kilometers above the planet's surface(Ref.2, p.45). This complex atmosphere is believed to contain heavier molecules than methane which was previously believed to make up the entire atmosphere(Ref.7, p.326). Other atmospheric properties which must be investigated include, measurements of temperatures and pressures at various altitudes and cloud characteristics (if present).

A second characteristic which needs investigation is the surface characteristics of the planet. Earth observations have shown the existence of polar ice caps at the poles of Pluto which are believed to be composed of methane ice (Ref.13, p29). This possibility along with other surface features need investigation.

Other areas of interest include, mass, shape, density, orbit characteristics and composition. By investigating these areas, we hope to gain improved knowledge of the planet Pluto.

3.2.2 Charon Characteristics

Pluto is believed to have only one orbiting natural satellite named Charon. The characteristics which need to be studied are relatively the same ones found in the previous section. One difference is that the amount of methane on Charon is believed to be much less than on Pluto. Charon is believed to be composed of water ice and not methane ice.

3.2.3 Planetary and Interplanetary Particles and Fields

One interesting area which falls under this category is the gravitational and magnetospheric interactions of the Pluto-Charon system. Charon is relatively large compared to Pluto. It is because of this that the Pluto-Charon system was thought to be one planet which led to incorrect measurements. There is no other planet-satellite system known so it seems very important to study these interactions.

Other areas shall include investigation in; charged particle environments, wave particle interaction, solar wind and cosmic rays.

The instrumentation used in most of these measurements is located on the probe's scientific boom which allows for measurements in the interplanetary environment as well as the planetary environment.

3.3 Pulse Experimental Package

The Pulse Experimental Package(PEP) will consist of five remote sensing instruments and four particle and field instruments and radio science. Each of these instruments is listed in Table 3.2. Also listed in this table are mass and power specifications. The total PEP weight is approximately 94.9 kg and the approximate power they consume is 80 W. The selection of these instruments was based on their ability to investigate the scientific objectives.

3.3.1 REMOTE SENSING INSTRUMENTS

IMAGING SCIENCE SUBSYSTEM

The Imaging Science Subsystem(ISS) was selected because it has a much higher resolution (1024 x 1024 pixels) than any of its predecessors(Ref.5, p.9). Many of the instrument's components are just improvements upon the camera systems of its ancestors. This instrument also offers data compression and storage which will be necessary because of the large amount of data that will be obtained during our flyby of the Pluto-Charon system since most of the investigation will be carried out at this time. The data rates of the ISS are selectable. They range from 6.2 kbps to 350 kbps(Ref.5, p.10).

The ISS offers the opportunity to view the Pluto-Charon system. The characteristics of Pluto and Charon will be investigated with the ISS. We also will be able to investigate

Table 3.3

NAC and WAC Optics

NAC

Type:	Ritchey Chretien with three field correctors
Focal Length:	2000 millimeters
Focal Ratio:	f/10.5
Spectral Range:	200-1100 nanometers
Resolution:	The resolution per pixel will be six microradians square.
Coverage:	The field of view will be 0.35 degrees square.

WAC

Type:	Refractor
Focal Length:	250 millimeters
Focal Ratio:	f/4.0
Spectral Range:	350-1100 nanometers
Resolution:	The resolution per pixel will be 48 microradians square.
Coverage:	The field of view will be 2.8 degrees square.

-Ref. 3, p. 8

the Pluto-Charon interactions and determine other areas that may be of interest in Plutonian Space.

This instrument, which is essentially the same as the ISS that will be flown on the Cassini and CRAF missions scheduled to be launched in 1995 and 1996, is composed of two cameras, a Narrow Angle Camera(NAC) and a Wide Angle Camera(WAC). The cameras will have a spectral range which is extended visible and they well operate at a temperature slightly below room temperature. The components of these two cameras include a dust cover, hood, optics, filter mechanism, shutter detector head and radiator. The dust covers are a method of protection for the optics which will be motor activated. The hood is designed to also protect the optics and reduce the glare. The optical parameters for both the NAC and the WAC are listed in Table 3.3(Ref.5, p.9).

The filter mechanism of the cameras was derived from the Hubble Space Telescope. Unlike Galileo's filter mechanism that had a maximum of seven positions, Pulse's filter mechanism has a maximum of 36 positions. The two filter wheels of the NAC and the WAC contain 22 filters and 14 filters respectively(Ref.5, p.10).

The shutter technology oriented from shutters on Voyager and Galileo. It consists of a dual blade focal plane which may operate in either direction. The lower limit on exposure time is .005 seconds and no limitation on the upper limit. One advantage of this system is that both shutters may be activated simultaneously (Ref.5, p.9).

The detector head of the ISS contains the Charge Coupled Device(CCD), driver, thermal control unit and signal chain circuits. This electronic module is common to both the NAC and the WAC. Other components of this module include: 1) a microcomputer 2) memory 3)power supplies 4) engineering sensors 5) image data multiplexer 6) square root processor 7) image memory 8) image data compression 9) bus interface unit(Ref.5, p.10).

The radiator of the ISS is responsible for cooling the CCD to temperatures approximately -80 degrees Celsius(Ref.5, p9).

NEAR INFRARED MAPPING SPECTROMETER

The Near Infrared Spectrometer(NIMS) is one of the instruments that is aboard the spacecraft Galileo. This instruments unique ability of combining spectroscopy and imagery in one instrument makes it a prime candidate for PEP. Another reason for its selection is that it can monitor both methane and water vapor which are believed to be present on Pluto and Charon respectively (Ref.8, p.207).

The objectives of NIMS fall into the first two scientific objectives. NIMS will be used for both the investigation of geological properties of both Pluto and Charon. NIMS will accomplish this objective by investigating surface features and surface composition through surface mapping and infrared spectral investigations.

NIMS will also investigate atmospherical properties. Goals of this investigation include information about atmospheric structure and composition. Also investigations about the existence of clouds, cloud properties and temperatures at various altitudes will also be conducted. Table 3.4 lists a summary of specifications for this instrument.

The NIMS will be placed on the scan platform. It is protected against contamination by covers and heaters. It also has a passive radioactive cooler which will keep the instrument at is operation temperature of 80 K(Ref.1, p.201).

PHOTOPOLARIMETER-RADIOMETER

Photopolarimeter-Radiometer(PPR) was also an instrument flown on the Galileo spacecraft. It was selected primarily because of ability to measure its intensity and linear polarization of scattered sunlight in the spectral region where methane strongly absorbs radiation(Ref.19, p.128). It is also unique because of the combination of three separate experiments it may conduct; photometry, polarimetry and radiometry.

The objectives of this instrument is as described above to measure the intensity and linear polarization of scattered sunlight in the narrow spectral bands.

Another objective of the PPS is the measurement of thermal infrared radiation. This may only be investigated if clouds do exist in the Plutonian atmosphere since the radiation is believed to be emitted primarily from cloud particles.

Some atmospheric properties well also be investigated. This experiment is mostly concerned with the particles in the atmosphere and their distribution.

Table 3.4

NIMS Instrument Characteristics

Angular Resolution:	0.5 mrad x 0.5 mrad
Angular Field:	10 mrad (20 pixels) x 0.5 mrad (1 pixel)
Spectral Range:	0.7 - 5.2 micrometers
Spectral Scan Time:	4-1/3 seconds (20 pixels, 204 wavelengths)
Telescope:	23 cm diameter f/3.5 Ritchey - Chretien wobbling secondary for spatial scan, 800 mm equivalent focal length
Spectrometer:	40 lines/mm plane-grating spectrometer, f/3.5 Dall Kirkham collimator f = 400 mm, f/1.86 wide-angle flat-field camera f = 210 mm
Detectors:	InSb (15), Si (2), discrete elements, quantum efficiencies = 70-80%, noise equivalent power = 10^{-14} watt, $D^* = 3 \times 10^{13}$ cm $\sqrt{Hz}$ watt^{-1}
Signal-to-Noise:	100:1 (0.075 albedo surface at 3 micrometers)
Mass:	18.0 kg
Power:	12 W (average), 13 W (peak)
Date Rate:	11.52 kbps
Data Encoding:	10 bits

-Ref. 1, p. 201

There are several different channels for the PPS the "polarimetry channels are centered at 4100, 6780, and 9450 and the photometry channels are centered at 6180, 6330, 6460, 7980, 8300, 8410, and 8920 angstroms. When the instrument is used for radiometry the infrared channels are centered below 4 micrometers at 17, 21, 27.5, and 37.5 micrometers, and above 42 micrometers." (Ref.19, p.129)

There are two operational modes, a cycle mode and a radiometry mode. The cycle mode rotates the filter wheel allowing each channel to transmit at least once every 18 seconds. The radiometry mode rotates the infrared filter wheel back and forth.

The PPS weighs 4.8 kg and has both a replacement heater and a sunshade as safety features(Ref.19, p.129).

ULTRAVIOLET SPECTROMETER

The ultraviolet spectrometer was selected for determining the composition and structure of the planet Pluto and its satellite Charon.

A secondary objective of this instrument is to determine the properties of the upper atmosphere. Although Pluto's atmosphere may not be as large as that of Jupiter, there is a possibility of molecular absorption features and auroral zone emissions that are believed to be common among planets with large atmospheres. Through airglow and occultation modes we hope to determine both the atmospheric structure and the atmospheric composition.

This Galilean successor will consist of a 250 mm-aperture

Cassegrain telescope, a 125 mm focal length Ebert-Fastie monochromator, three detectors and control logic. The UVS weighs approximately 4 kg and consumes 5.33 W. The wavelengths covered by the UVS range from 1100 to 1400 angstroms(Ref.19, pp.130-131).

The UVS also has flexibility. It may take data at a fixed wavelength or it may change the wavelength every 0.0007 second. It is not limited to these two modes, however. Other variations may be programmed into the microprocessor of the UVS (Ref.19, p.131).

3.3.2 PARTICLE AND FIELD INSTRUMENTS

MAGNETOMETERS

The magnetometers that were selected for PEP are actually the same magnetometers used aboard the Voyagers. They were selected because of their ability to measure fields ranging from 0.006 gamma to 20 G(Ref.4, p235). This wide range of field measurements will be needed to measure the fields in both the Plutonian and interplanetary environments. The fact that the PULSE probe is three-axis stabilized, like Voyager, also gives reason for this selection.

The magnetometers that have been selected are two Low Field Magnetometers(LFM) and two High Field Magnetometers(HFM). This redundancy makes the system reliable in the event that one of the magnetometers does not function properly. The magnetometers purpose is to study the planetary and interplanetary particles and fields. These objectives are described as follows:

1) Investigate Pluto-Charon magnetospheric interactions.
2) Measure the magnetic field of Pluto and Charon.
3) Measure interplanetary magnetic fields
4) Determine magnetospheric interactions with solar wind,
cosmic rays and plasma waves.
5) Use observations to make further observations.
6) Search for interaction between interplanetary and interstellar media.

The LFM and the HFM are located on the particle and field boom. The placement of these magnetometers will be proportionately the same as the ones on the Voyager missions. There will be one LFM located at the outboard end of the boom and the other LFM will be placed approximately at the center of the boom. The two HFM will be located near the inboard end of the boom approximately one meter apart. This placement allow for some measurement correction factors due to the spacecraft's magnetic field(Ref.4, p.247).

The range of the measurements as state earlier is fairly large. The LFM range is ±8.8 gamma to ±0.50 G and the HFM range is ±0.50 G to ±20 G with uncertainties of ±2.2 milligamma to ±12.2 gamma and ±12.2 gamma to ±488 gamma respectively. This total ±20 G range has a 12 bit digital resolution(Ref.4, p.236).

As the probe increases its distance from the sun, the data rate will not vary greatly because of the data compaction modes of the instrument(Ref.4, p254).

COSMIC RAY DETECTOR SYSTEM

Like the magnetometers of the PEP, The Cosmic Ray Detector

System(CRS) selected for PEP has also flown on the Voyager missions. This instrument was selected because Earth-based observations show that something is blocking the light during Pluto's occultation. There are beliefs that this "extinction layer" is produced by particles which originated from cosmic rays(Ref.13, p.29). Therefore the CRS investigation may enhance our knowledge of both cosmic rays and the components of the Plutonian atmosphere.

The CRS objectives fall in the category of planetary and interplanetary particles and fields. These objectives may be almost exactly compared to those of the Voyager CRS objectives. There only difference is the planet that is being targeted. Below is a list of the objectives of the Voyager mission from the Flight Science Office Science and Systems Handbook with the appropriate modifications for the Pluto mission.

1) Measure the energy spectrum of electrons 3-110 MeV.
2) Measure the energy spectra and elemental composition of all cosmic ray nuclei from H through Fe over an energy range from approximately 1-500 MeV/nuc.
3) Provide information on the energy content, origin, acceleration process, life history and dynamics of cosmic rays in the galaxy and contribute to an understanding of the nucleosynthesis of elements in cosmic ray sources.
4) To provide information on the transport of cosmic rays, Plutonian electrons and low energy particles over an extended region of interplanetary space.
5) Measure the three-dimensional streaming patterns of the nuclei from H through Fe and electrons over an extended range.
6) Measure particle charge composition of the magnetosphere of Pluto and Charon(Ref.17, p4.1)

One may say that these objectives, inherited from the Voyagers, are still of great importance to the scientific community.

ORIGINAL PAGE IS
OF POOR QUALITY

The CRS is composed of three systems; the High Energy Telescope System, the Low Energy Telescope System and the Electron Telescope System. These three systems share some common electronics and are responsible for the above objectives. The nuclei charge and energy spectra may be determined by these instruments for elements with atomic numbers from 1 to 30 and energy ranges of 1 MeV to 500 MeV for H and 2.5 MeV to 500 MeV for Fe. For isotopes the range of atomic numbers is 1 to 8 with an energy range of 2 MeV/nuc. to 75 MeV/nuc. Finally, the range of atomic numbers of anisotropies is 1 to 26 with an energy range of 1 MeV to 150 MeV for H, 2.7 MeV to 500 MeV for Fe and 3 to 10 MeV for electrons (Ref.4, p.365).

PLASMA INSTRUMENT

The Plasma instrument(PLS) that has been selected was flown aboard the Galileo Spacecraft. It was selected because of its energy/unit charge and the decreased temporal resolutions for obtaining electron and positive ion spectra. The plasma instruments of the Voyagers and the Pioneers don't even approach the values of the PLS.

The objectives of this mission are also of the particle and field type. These objectives include measurements of the plasma properties in solar wind, assessments of composition, energy, intensities and three-dimensional distribution of low energy particles.

The PLS is composed of the following:

1) Two electrostatic analyzers that measure the energy/unit charge of electrons and positive ions.

2) Seven sensors that determine electron intensities.
3) Seven sensors that determine positive ion intensities.
4) Three mass spectrometers that determine the composition of ions(Ref.19, p.133).

The PLS capabilities range from 1 Vto 50,000 V in 64 different passbands. The PLS also contains software which permits ground command alterations to the instruments commands. The instrument weighs 12 kg and will be mounted on the science boom of the PULSE probe(Ref.19, pp.133-135).

ENERGETIC PARTICLE DETECTOR

Another instrument selected from the Galilean payload is the Energetic Particle Detector(EPD). It was selected because of the need for measurement of high energy particles in the magnetospheres of Pluto, Charon and interplanetary space. Although the PULSE probe is three-axis stabilized, we should still be able to obtain a great deal of data about the high energy electrons, protons and heavy ions even without sweeping motions.

The EPD is made up of two subsystems, a Low Energy Magnetospheric Measuring System(LEMMS) and a Composition Measuring System(CMS), formed by two separate telescopes(Ref.19, p.136).

The LEMMS consists of two components. The first component is an ion telescope with two solid-state detectors. One detector, the low field detector covers an energy range of 0.02 MeV to 3.4 MeV. The other detector will be used for the definition of additional electron, proton, and alpha particle channels. The second component of the LEMMS is a magnetic electron spectrometer with two detector pairs. These detector pairs span a range of 0.015 MeV to 0.20 MeV and 0.10 MeV to 1.0 MeV(Ref.19, p.136).

The CMS components will be used for the measurement of composition, energy spectra and pitch angle distributions of the high energy ions. These components are the CMS telescope and nine detectors(Ref.19, p.136).

The EPD weighs 9 kg and will also be located on the science boom(Ref.19, p.6).

PLASMA WAVE SUBSYSTEM

The last particle and field instrument is the Plasma Wave Subsystem(PWS). The PWS was selected because of the importance of plasma wave investigations.

These investigations include wave particle interactions and their effects on the Pluto-Charon system and measurements of spectral characteristics of electric and magnetic fields in the range of 5 Hz to 5.65 MHz. We will also be able to distinguish the difference between electrostatic and electromagnetic waves(Ref19, p.137).

There are two sensors of the PWS. The first is a 6.6 meter electric dipole antenna which has two tapered graphite epoxy elements mounted at the end of the magnetometer boom. The other sensor is a search coil magnetic antenna. This antenna consists of two high-permeability rods, 26.6 and 27.5 cm long. The low frequency search coil has a winding of 50,000 turns of 0,07 mm diameter copper wire and a frequency range of 10Hz to 3.5 kHz. This search coil must be mounted parallel to the electric antenna. The high frequency antenna has a winding of 2,000 turns of 0.14 mm copper wire and a frequency range of 1 Hz to 50 kHz. This search coil must be mounted perpendicular to the electric antenna. There will also be a preamplifier mounted near the search coil to provide a low impedance to the electronics(Ref.19, p.136).

The processing of the signal received from the sensors may be processed by a low-frequency spectrum analyzer, a medium-frequency spectrum analyzer, a high-frequency spectrum analyzer and a wideband waveform receiver. The fastest measurements are provide by the wide band waveform receiver(Ref.19, pp.136-137).

3.4 CONCLUDING REMARKS

The Objectives in this subsystem report are by no means the only investigations that will be conducted. There are indeed some that were not mentioned and some that will not materialize until a probe visits Plutonian space. The purpose of this mission is to observe as much as possible so as to enhance our knowledge for further scientific investigations.

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4.0 Introduction to command, control, and communication

The command, control, and communication subsystem has several design requirements which include:

1) minimization of cost and weight
2) maximization of performance of reliability, performance, and simplicity
3) use of off-the-shelf hardware
4) use of technology before 2000
5) application of AI, if applicable
6) sufficient life time to carry out the mission

The priority that overshadows all of them is cutting the cost of the mission. As far as incorporating new technology into PULSE, we are taking a conservative approach. Proven designs will be chosen over new technology, except in the case where it would be more cost effective to use the latter. When possible, past deep space probes will be used as a prototype due to reliability and cost requirements.

4.1 Antenna System

Reliability is the dominating factor when discussing antennas. Voyager 2 and Galileo will be used as the prototype for this subsystem due to the fact that proven techniques enhance reliability and lower the overall cost of the vehicle. A high-gain circular parabolic antenna will be used because this shape optimizes the gain. A low-gain antenna will be included mostly for communication when near earth for attitude articulation and control reasons, since the high gain antenna can not be used these ranges.

4.1.1 High-gain antenna

The high-gain antenna (HGA) meets all of the requirements stated in the RFP. HGA's are the most cost efficient antennas because they use off- the- shelf hardware. They are reliable and their performance is well known because they were used in many previous spacecraft and are based on already proven technology. This antenna was chosen because it meets all of the applicable requirements.

4.1.2 HGA trade- offs

The most important trade- off in HGA's is the power-gain tradeoff. Gain is increased as the antenna size is increased, this also result in a higher weight. If more power is needed the weight also increases because the weight of the RTG's must be greater. This is accompanied by the requirement of minimizing the weight of the antenna. The maximum of power- gain trade- off occurs when the product results in minimum weight.

4.1.3 A Look at Laser Communication

Optical communication could result in 47 bps from 50 AU from a mass of one kilogram. There are many reasons that this technology cannot be justified given the requirements from the RFP. Optical communication is in the high- risk department as of now because it has not been deep space tested yet. Plans for testing are planned but it is doubtful optical communication will be ready for deep space missions before the year 2000. This antenna would also require that a 20 m receiving antenna be put in orbit, since optical communications have a severe limiting factor of weather dependence.

4.1.4 Size of High-gain Antenna

The size of the high-gain antenna is going to be 2.5 meters in diameter. This is the maximum size that the launch vehicle will allow. This is smaller than either Voyager or Galileo, which are 3.66 and 4.8 meters in diameter consecutively. This decrease in size can be accounted for in several different ways including increase of gain in the antenna, improvements in the Deep Space Network (DSN), and improvements in the encoding and decoding of data.

4.1.4.1 DSN

The DSN applies the technique of antenna arraying. It includes many large antennas from all over the world.

LOCATION	DISH SIZE	X-BAND REC´V
GOLDSTONE	34m	YES
	70m	YES
	34m	YES
V.L.A.	27x 52m	YES
CANBERRA	34m	YES
	70m	YES
	34m	YES
USUDA	64m	NO
PARKES	64m	YES
MADRID	34m	YES
	70m	YES
	34m	YES

Possible improvements to this network include changing the Usuda antenna so it is capable of X- band reception. Increasing the size of the 64 m antennas to 70 m. Adding a 34 m antenna at the Parkes and Usuda location would add 1.1 db each. General Electric has suggested that the masers be replaced by high- electron- mobility transistors, which would cost a third as much to operate and a quarter of the implimentation cost. These improvements could led to 3-4 db increase in gain.

4.1.4.2 Encoders and Modulators

The effectiveness of digital satillite communications systems (DSCS) will increase when well chosen modulation and noise- immune encoding methods are used. The PSK-4-CC was found to to be a good method. Both the frequency effectiveness and energy can be increased. Power gains may reach 5 db and specific rates can increase by a factor of 1.5. From a costing side, increasing the efficiency of the encoder is less expensive than increasing antenna size or transmitted power, or increasing the receiver noise sensitivity.

4.1.5 Amplifier

The amplifier used will very from the one in Voyager 2, but will be similar to the one used for the generic Mariner Mark 2 (MM2) design. This design includes the use of gallium arsenide field-effect transistors in the amplifier to produce an output of 5.6 W. This value could be raised to about 10 W with only minor modifications. This application of solid state electronics would cost less than half that of the system used in the Voyagers which featured traveling-wave-tube-based amplifiers.

4.1.6 Radio-frequency Subsystem

PULSE's high-gain antenna will maintain communication with Earth in only X- band, as in the case of CRAF. S- band communication was used in the Voyagers because not all ground stations could not handle X- band when they were launched. Now, all stations except the Japan based antenna are capable of X- band communication. X- band offers better range and range- rate measurements, and greater immunity to charged particle interference. Using only one band simpifies the ground system and lowers the operational costs.

4.2 On- board Computers

Radiation- hardened versions of widely available microprocessors and integrated- circuit chips supported by well-known software development tools. Handling of scientific data during and after the mission must make use of the latest technology.

4.2.1 Lag in Technology

The computer industry is one of the most rapidly developing industries. There has been a problem with computer systems in past spacecraft due to the lag in technology because of this rapid development. This is difficult to avoid because of the time delay between deciding on a system and the actual launch date.

4.2.2 Performance Characteristics

The PULSE probe will be outdated by the time it is launched, as in the case of all spacecraft, but on- board computers need to be selected about five years in advance to develop, test, and integrate the spacecraft subsystems. A schedule and summary of major features of the PULSE computer system are listed below.

Launch date	2003
Year computer selection made	1993
Year commercially available	1990
Difference in launch and avalable	13
Microprocessor	32 bit
Performance	4 MIPS
RAM	4000 kbytes possible

4.2.3 Space Qualification of Computers

The problem with spacecraft computers is that they must be able to withstand radiation and the bombardment of high-energy particles, and operate in a highly reliable manner. NASA, Defense, and the Department of Energy are working to develop and deploy space qualified computers.

There are several space qualified computers. Sandia National Laboratory is developing a set of advanced 32- bit and 16- bit microprocessors called the SA 3300 family. The microprocessor and its associated computer hardware should be available in about four years. There is also a generic version of the 32- bit processor RH32 which will be fully developed soon.

4.2.4 Computer trade- offs

Because of size, weight, and power limitations on-board computers must be small in size, lightweight, and have low power requirements. Selecting more advanced computers for the spacecraft can result in higher development costs, but the overall result is lower overall life- cycle costs of space missions through lower software development and maintenance costs. This can be further decreased when a universal higher level languages are approved for space programs. The Department of Defense approved Ada recently. The advantage for this standardization is lower cost, lower development risks, shorter delivery schedules and ease of maintenance. To date, assembly language source coding has been used for spacecraft data processing. Sufficient support software should be available by the time PULSE is launched. The emphasis will turn from hardware to software to control the spacecraft. By putting all the sophisticated logic in software, much less hardware is needed and designers have the flexibility of reprogrammability.

4.2.5 Problem with Galileo

NASA used a RCA 1802 8- bit microprocessor which caused problems due to the limited capabilities. Its relative low speed and its limited memory increased cost because of problems with writing efficiency and maintainable software. The 32- bit processor in PULSE will allow expanded mission objectives such as acquiring and relaying more pictures faster, and allowing more autonomous operations. While scientific objectives could be reached with a less modern computer, lower cost and risks encourage its use.

4.2.6 Data Management Systems (DMS)

The DMS must regulate power management, command and telemetry, thermal regulation, and antenna control. centralization of the DMS ensures command prioritization and synchronization of resources. Using separate microprocessors and spares can result in power, weight, and code complexity to provide the necessary redundancy. The DMS may make use of a internally redundant Intel 80386 for data processing and automatic control purposes. The only problem is that it is not radiation hardened yet and may not be by the year 2000. If it is not a back-up option would be a 32- bit radiation hardened microprocessor combined with a direct memory access chip that simplifies software which is being developed by JPL.

The DMS will be similar to the ESA probe ISPM include a Central Terminal Unit (CTU), Remote Terminal Unit (RTU), Command Decoder, and data storage (a tape recorder or hard drive). The CTU controls the automatic functions and operations. The main tasks will be performed on the Intel 80386 microcomputer. The software governing articulation and control is based on the Ada language. The CTU contains a fault detector which will switch to redundant units when problems arise. The command detector that will be used is the NASA standard which is upgraded from the one used in Galileo.

4.3 Conclusion

The most important features of this subsystem is the 2.5 m high- gain antenna which will communicate with the Deep Space Network at a distance of around 33 AUs with x- band uplink and downlink and the centralized Data Management System which utilizes the Intel 80386 computer, and the Ada language for software applications.

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5. STRUCTURE

5.1 Requirements to be met by the structure:

The structure has the objective to support all other subsystems and carry them out to Pluto safely. It has to protect them from destruction or damage and also from influences which might affect the performance of those subsystems. In this context the following requirements were derived from the RFP.

use no materials available after 1990

lifetime long enough, with a safety margin

weight and cost optimization

stress reliability

stress simplicity

stress low cost

nothing should preclude other missions

interface to the launch vehicle

if necessary, on orbit assembly should be minimized

5.2 Shape and Configuration:

5.2.1 Grouping:

The structure of PULSE has to support all subsystems and meet all the different requirements from those systems. In order to comply with conflicting requirements, groups of subsystems with similar requirements have to be placed together. This subsystem grouping yielded 4 major areas with different necessary attributes:

The main body :

Requirements: provide thermal environment
support mass
radiation shielding
micrometeoroid protection
withstand launch forces

Subsystems: Communication electronics
Control electronics
Data storage
Gyroscopes
Power conditioning equipment
Fuel pumps and lines

To meet these requirements the subsystems have to be encased in a shell which will protect the inside from micrometeoroids, radiation, will not yield due to the launch forces and provide a sufficient insulation against heat loss. Conflicting requirements are here low cost and low weight against high protection and strength. Desirable is also good damping of vibrations during take off to protect the electronics from mechanical damage.

The science boom :

Requirements:	negligible magnetic and electric interference support mass provide thermal environment micrometeoroid protection
Subsystems:	magnetic field instruments particle detectors

The predominant point in this group is, that the science instruments have to be able to measure an as much as possible undisturbed environment. To keep disturbance by the electronics on board the probe as low as possible, those instruments have to be away from the spacecraft. Even though micrometeoroid protection is necessary, shielding is not feasible since that would shield off the fields to be measured also. The same applies for the heating. On one hand the electronics needs to be kept at an operating temperature, but on the other hand, heaters would create a disturbance. For these reasons, the instruments have to provide these measures themselves.

The science platform:

Requirements:	Pointability and good field of view support mass micrometeoroid protection provide thermal environment pointability
Subsystems:	Science instruments (cameras,infrared spectrometer)

Other science instruments require less shielding than the field and particle instruments. For this reason they can be mounted on the main body and micrometeoroid protection and heating can be supplied by the structure. In addition to the control electronic housed in the main body these instruments needs to be pointable and they have to have a good field of vision. This is accomplished by separating them from the main body and mounting them on a movable platform on top of the main body. To ensure the micrometeoroid protection, a steel canopy is placed over the platform. Steel has been chosen to maximize the protection since the science instruments are the essential parts of this mission. During the cruise phase it will be closed and only when PULSE approaches Pluto it tilts open. The platform will be turnable by 360 degrees and tiltable by +- 15 degrees. These values ensure that a large area can be scanned by the mounted instruments.

The power boom:

Requirements: micrometeoroid protection
allow heat radiation
support mass

Subsystem: RTG

RTG's radiate a large amount of unwanted radiation which would have a negative influence on the performance of electronic equipment, this radiation has to be kept away from those instruments. It would require heavy shielding to protect the computers which would interfere with the requirement of low weight. It also would affect the necessary heat radiation of the RTG's. Thus the RTG's have to be moved away from the main body. This yields now two booms which can be spaced by 180 degrees to enhance symmetry and maximize the distance between the sensitive science instrumentation and the high radiation of the RTG's. The spacecraft body also functions as a shield. The science platform will not be operational during the cruise phase. During the flyby, the open steel canopy will be tilted in the direction to the RTG's to provide shielding.

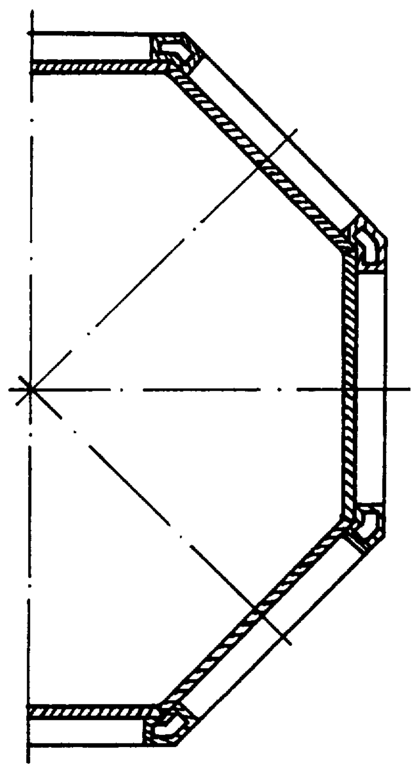
Other subsystems:

The remaining subsystems are the antenna, the propulsion tanks and the startracker and sun sensor. The predominant requirement for the antenna is, that it has to be pointed to Earth at all times. Additionally the antenna is required to function as an adapter interface with the launch vehicle. This yields, that the antenna is firmly mounted on the main body to provide the necessary support. Thus the whole body of the spacecraft will be pointed at earth.

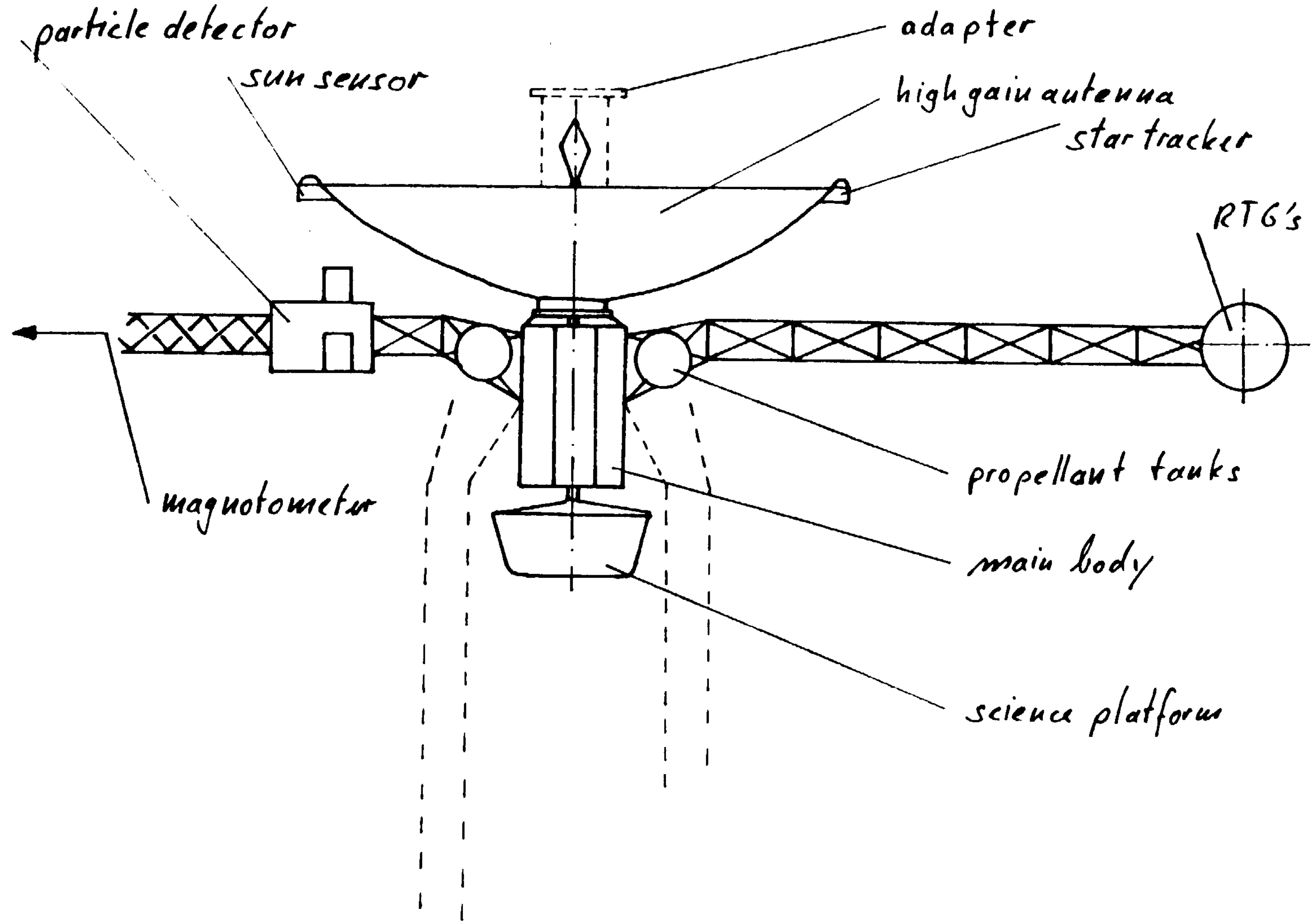
The propellant tanks will be bought from stock and placed next to the main body on both sides of the boom structure. This will limit the volume needed for the main body and thus decrease the weight. There will be four propellant tanks and the their steel body will provide a sufficient protection against micrometeoroids.

The startracker and the sun sensor need a good field of vision to be able to scan a large area. This is accomplished by placing them on the rim of the parabolic antenna. Both have similar pointing requirements, and since the difference in angles to the sun and the earth is maximal 12 degrees in the periphery of our sun system the instruments have to provide only a small correction to their pointing. Here they also have a large angle available where no obstacles block their field of vision.

Cross Section of main body



The PULSE Space Probe



5.2.2 Shape determination

The main driver when determining the shape of the main body, is the prevention of heat loss to space. An important variable there is the surface. The smaller the surface, the smaller the heat loss. Therefore I considered shapes which allow me to have a large volume but also have a small surface area. Obviously the sphere has the highest volume to surface ratio (V/S ratio) but production and interface problems make the sphere less desirable to be used on PULSE. I then considered the cylinder. It has a smaller V/S ratio, but provides two flat interface surfaces. Looking at the amount of equipment to be mounted inside the hull it is apparent, that this is not enough. Adapters need to be installed to fit the instruments to the curved surfaces. This would increase the weight of the structure and complicate the manufacturing. From these considerations I propose a regular octagon as the shape of the main body. It has still a high V/S ratio but has flat sides so the instruments can easily be mounted.
From the volume required I derived the design sizes. This yielded a diameter of 0.5 m and a height of 0.8 m.

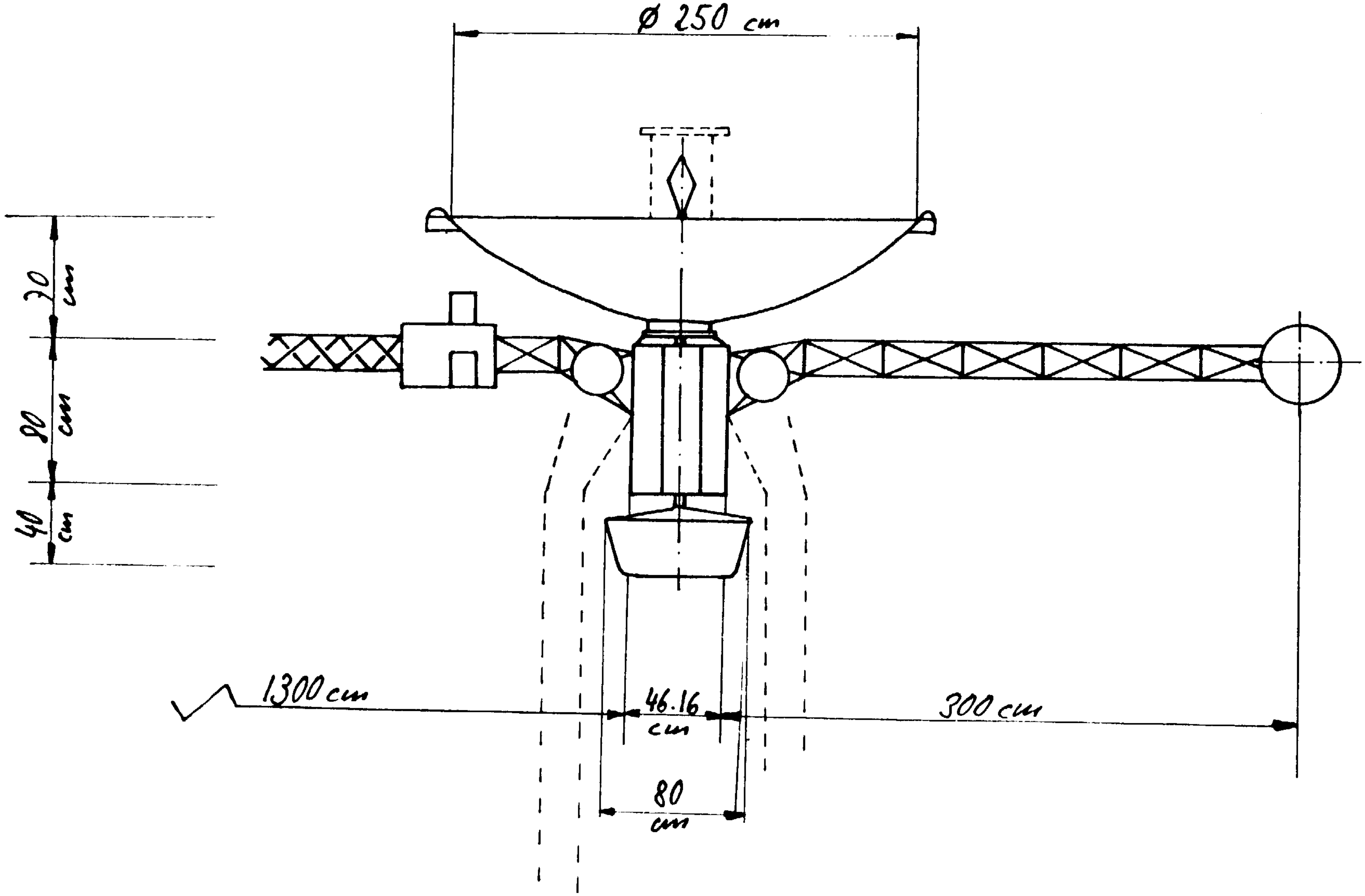
5.2.3. Configuration:

Due to the requirements of having both RTG's and highly sensitive particle and field instruments on the same craft, it is necessary to separate them as far as possible. For this reason booms need to be employed. I propose two booms, one carrying the two RTG's and the other all the particle and field sensors. This enables a 180 degrees separation which gives the maximum separation distance. This way the main body also acts as a shield in between. Since even the on board electronics interfere with those sensors, the science boom needs to be considerably longer than the power boom. Only 3 m are necessary for the power boom this allows the downward folded boom to fit in the launch vehicle in it full length. The science boom, which requires a length of 10.6 m needs to be partially retractable. This retraction technique can be directly inheritated from the Galileo spacecraft.

The antenna will be firmly mounted on top of the main body so that its center section can support the adapter to the launch vehicle. I also considered making the antenna pointable. This would decrease the attitude correction maneuvers and thus reduce the necessary amount of propellant. Added weight and complexity due to the pointing mechanism and compatibility problems with the launch vehicle discard this option. A pointing mechanism would not be able to provide a stiff support when placing the adapter on the antenna. A complex design is necessary to comply with both, the pointability and the stiffness during launch. Placing the adapter on the other side of the craft requires a very large adapter because it has to give room to the booms and using the booms is not feasible because they, as the pointing mechanism are not stiff enough to firmly support the probe during launch.

Since the remote sensing instruments need to be pointed at the object of interest and the antenna needs to be pointed at earth, a pointing mechanism is necessary for the science platform which will house the remote sensing equipment. These can than be

The PULSE Space Probe



pointed independently from the main body. During the cruise phase these instruments are not used and to protect them a steel canopy is placed over them. This canopy will tilt open when the instruments are operational.

5.3. Material selection:

To perform the material selection I gathered as much information from different sources as possible and incorporated them into the following table.

PROPERTIES:

Property	Al	Be	Mg	Ti	Kevlar	Steel	Unit
Density	2.8	1.85	1.74	4.5	1.9	7.87	g/cm^3
Yield str.	500	415	103	830	1600	1800	MPa
machinability	ex.	poor	ex.	good	poor	good	
weldability	good	poor	ex.	good	none	ok	
handling	ex.	poor	ok	ex.	poor	ex.	
cost	low	high	low	mod.	high	low	
corrosion resistance	ex.	ok	poor	ex.	ok	ex.	

I then awarded points for their properties on the scale of 0 through 100 according to the desirability of the properties.

POINTS:

Property	Al	Be	Mg	Ti	Kevlar	Steel	weight
Density	72	81.5	82.6	55	81	21.3	0.55
Yield str.	25	20.75	5.15	41.5	80	90	0.1
machinability	100	40	100	80	40	80	0.1
weldability	80	40	100	80	0	60	0.075
handling	100	40	60	100	40	100	0.05
cost	100	0	100	60	0	100	0.1
corrosion resistance	100	60	40	100	60	100	0.025
Sum :	577	282.2	487.7	516.5	301	551.3	1

The final evaluation is based on the points received and a weighing factor which allows to stress more important properties over less important ones.

EVALUATION:

Property	Al	Be	Mg	Ti	Kevlar	Steel	weight
Density	39.6	44.82	**45.43**	30.25	44.55	11.715	0.55
Yield str.	2.5	2.075	**0.515**	4.15	8	9	0.1
machinability	10	4	**10**	8	4	8	0.1
weldability	6	3	**7.5**	6	0	4.5	0.075
handling	5	2	**3**	5	2	5	0.05
cost	10	0	**10**	6	0	10	0.1
corrosion resistance	2.5	1.5	**1**	2.5	1.5	2.5	0.025
Sum :	75.6	57.4	**77.44**	61.9	60.05	50.715	1

Selection made: Magnesium

Legend:

Points	synonym
100	ex. or low
80	good
60	ok or mod.
40	poor
20	bad
0	none or high

Formulas used: For density : Points = 100 - density/10

=> density = 0 -> 100 Points
=> density =10 -> 0 Points

For yield strength : Points = Ys / 20

=> Ys = 2000 -> 100 Points
=> Ys = 0 -> 0 Points

5.4. Calculation of required wall thickness for micrometeoroid protection.

Material proposed:

Magnesium

Constants:

meteoroid mass,M :	0.1 g
meteoroid velocity,V :	25 km/s
meteoroid density,roh :	0.5 g/cm^3
mat. constant for Al :	0.06 (from reference)
mat. constant for Mg,K :	0.08 (estimated)
Density of Mg,RMG :	1.74 g/cm^3
Yield strength,YS :	22000 lbf/in^2

Derived Values:

meteoroid diameter,D : 0.725566 cm
(spherical meteoroid shape assumed)
first sheet thickness,T1 : 0.072556 cm
(T1/D=0.1 requ. by Formula)

Variable:

spacing,S : 2 cm

Formula : (for double sheet penetration)

t = K*roh^0.15*M^.35*V/S^0.*(70000/YS)
t = 1.015542 cm

Summary :

First sheet thickness,T1 :	0.072556 cm
Second sheet thickness,t :	1.015542 cm
Spacing,S :	2 cm

Protects from 0.1 g micrometeoroid at average speed.

Design sizes :

First sheet thickness,T1 :	0.2 cm
Second sheet thickness,t :	0.9 cm
Spacing,S :	2 cm

5.5. Mass estimation from design and sheet thickness:

Constants:

First sheet thickness,t1 :	0.2 cm
Second sheet thickness,t :	0.9 cm
Lid thickness,tl :	1 cm
Density of Mg,roh :	1.74 g/cm^3
Area of spar,Asp :	4.1 cm^2

Variables:

Height,h :	80 cm
Diameter,d :	50 cm

Formulas:

Panel length,s :	s = d/2 * (2-2^0.5)^0.5
	s = 19.13417 cm
Panel area,Ap :	Ap = 8 * s * (t1+t)
	Ap = 168.3807 cm^2
Spar area,As :	As = 8 * Asp
	As = 32.8 cm^2
tot. cross sect. area,Ac :	Ac = As + Ap
	Ac = 201.1807 cm^2
Lid area,Al :	Al = D^2 * 2^.5 / 2
	Al = 1767.766 cm^2
Lid volume,Vl :	Vl = 2 * Al * tl
	Vl = 3535.533 cm^3
Trunk volume,Vt :	Vt = Ac * h
	Vt = 16094.45 cm^3
total Volume,V :	V = Vt + Vl
	V = 19629.99 cm^3

Total weight of the main body structure:

M = 34.16 kg

5.6. Production techniques required:

The magnesium side panels can be bought from stock, cut and welded to the spars. The magnesium spars need to be extruded. The main body lids and the base of the science platform have to be casted. The steel canopy has to be produced by deep drawing and then weld the second sheet onto it to enhance the micrometeoroid protection. The boom struts can be bought from stock and then assembled.

All these techiques are well known and readily avalible today. Any new developements can be incorporated at a later point to improve the performance of the craft.

5.7 References:

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Propulsion

Numerous factors must be considered in selecting propellants and propulsion systems for space missions. One of the more general characteristics is performance, in terms of both specific impulse and hardware mass. Final selection must depend on tradeoffs between several of the major competing selection criteria: for example performance, reliability and cost.

The first decision to make was what launch vehicle the Pulse probe would be launched on. After evaluation of all of the United States vehicles and some International launch vehicles, it was found that the four best choices for this mission were the U.S. Space Shuttle, the Ariane IV, the Titan IV Centaur G Prime, and the Titan IV IUS. This primary trade study was based on the mass that each vehicle could be place into a geostationary transfer orbit. The United States Space shuttle was ruled out because of the higher cost for a non-expendable launch vehicle.

After this preliminary study a more in depth study was performed on the Ariane IV and the Titan IV configurations. Using the equations from Conway (Ref. 4), a comparison was made between the three launch vehicles on the basis of payload ratio, propellant mass and total mass, given a delta-v and a payload mass (Figures 6.1, 6.2, 6.3, 6.4, 6.5). The conclusion reached was that the Ariane IV launch vehicle was the best selection in all comparisons. The Launch Specifications for the Ariane IV are given in the appendix.

The fuel used for each stage of the Ariane vehicle will be the specified fuel in the launch specifications in the appendix. In these specifications one will find that the diameter of the upper stage is 2.59 meters in diameter which is sufficient for the largest diameter of our spacecraft which allow the antenna to fit in uncollapsed.

Fig.6.1

Subsystem Masses

System and components	Number of components (1+redundancy)	Weight (kg)
Science	1	94.9
Telecommunications		75.46
Control	2	22.73
Receiver	4	14.55
Amplifier	4	3.64
Data handling	2	16.36
Data storage	2	18.18
Spacecraft control		38.17
Computer and sequencer	2	10.91
Sun sensors	2	5.45
Canopus tracker	2	5.45
Gyros	2	5.45
Scan control and planet sensor	1	10.91
Electrical power		121.41
RTG's	1	44.4
Conditioning and control	2	45.45
Cabling	1	31.82
Structure and mechanical		290.46
Bus	1	150
Parabolic antenna	1	9.1
Temperature control	1	11.36
Trajectory correction propulsion		120
Total spacecraft weight		620.4
Launch vehicle adapter		50
Total injected weight		670.4

Fig.6.2

Launch Specifications

Variables	Ariane IV	Titan IV Centaur G Prime	Titan IV IUS
thrust1 [N]	204318.20	72715000.00	72715000.00
thrust2 [N]	40227.30	23636.40	23626.40
thrust3 [N]	3181.80	7500.00	13840.90
thrust (total) [N]	247727.30	72746136.40	72752467.30
c1 [km/s]	3038.00	2989.00	2989.00
c2 [km/s]	3136.00	3136.00	3136.00
c3 [km/s]	3528.00	3528.00	2842.00
c (total) [km/s]	9702.00	9653.00	8967.00
R1	2.56	2.46	3.45
R2	2.13	2.19	2.88
R3	2.90	2.95	1.99
R (total)	7.58	7.61	8.32
Ms1 [kg]	786.02	779.84	1210.00
Ms2 [kg]	334.52	366.74	372.91
Ms3 [kg]	125.11	129.82	57.84
Ms (total) [kg]	1245.64	1276.39	1640.75
Mp1 [kg]	10510.00	10420.00	16180.00
Mp2 [kg]	3161.00	3465.00	3524.00
Mp3 [kg]	1672.00	1735.00	773.20
Mp (total) [kg]	15343.00	15620.00	20477.20
Mo [kg]	17260.00	17570.00	22790.00
lambda 1	0.53	0.57	0.31
lambda 2	0.71	0.66	0.39
lambda 3	0.37	0.36	0.81
lambda (total)	1.61	1.59	1.50

Fig.6.3

Payload Mass Ratio

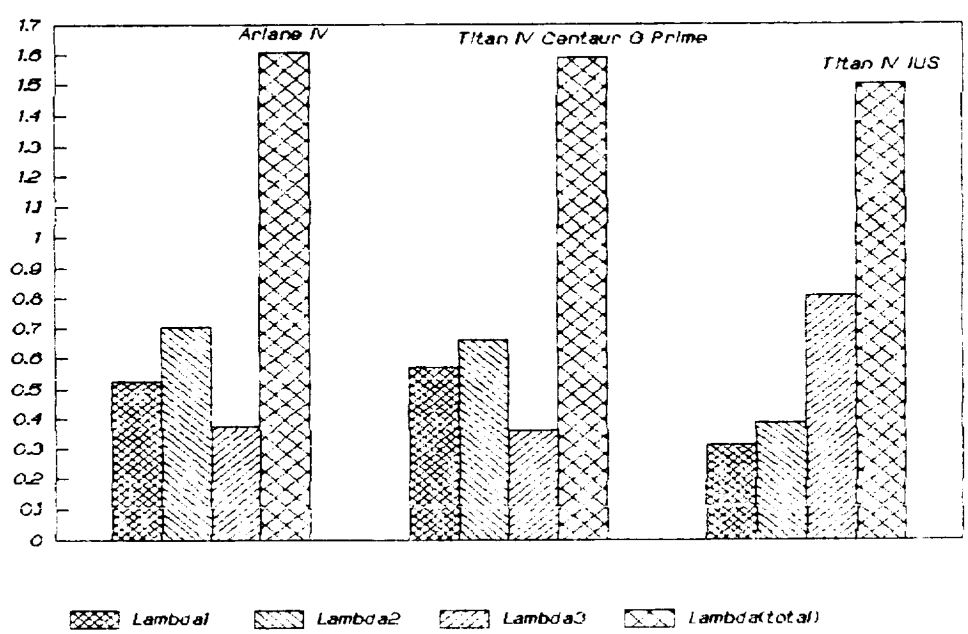


Fig.6.4

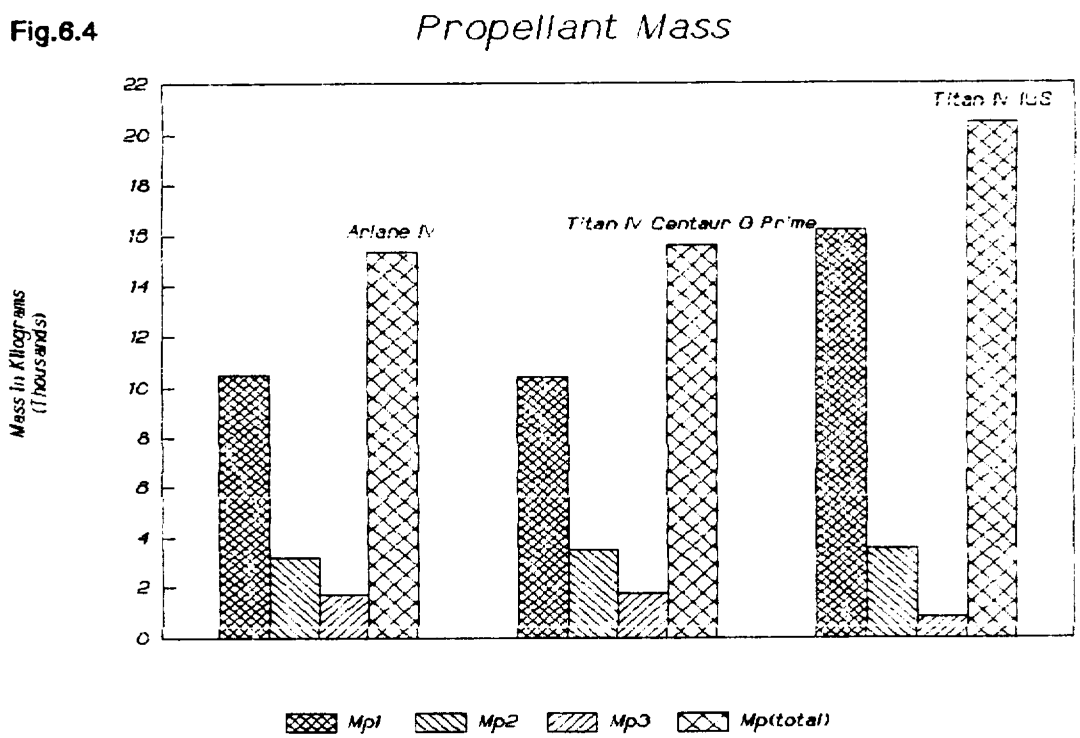
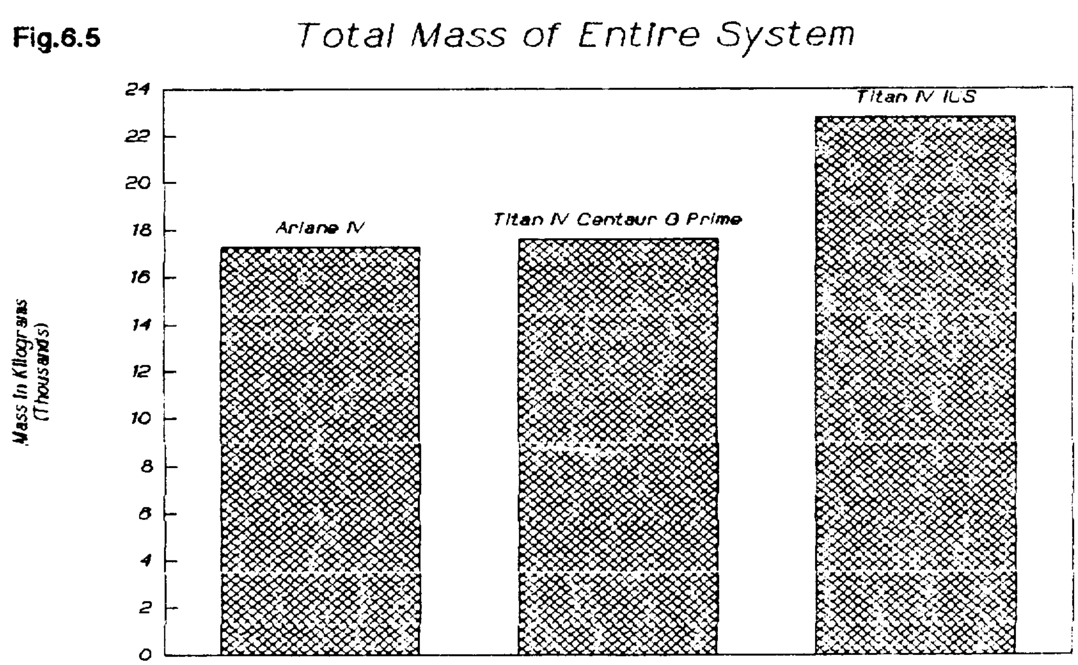

Propellant Mass
Mass in Kilograms (Thousands)
0
2
4
6
8
10
12
14
16
18
20
22
Ariane IV
Titan IV Centaur G Prime
Titan IV IUS
Mp1
Mp2
Mp3
Mp(total)

Fig.6.5


Total Mass of Entire System
Mass in Kilograms (Thousands)
0
2
4
6
8
10
12
14
16
18
20
22
24
Ariane IV
Titan IV Centaur G Prime
Titan IV IUS

Power System

The operational capabilities of a space vehicle is dependent upon an adequate supply of power. This power is necessary for communications, guidance, control, and operation of sensors or scientific instrumentation.

When trying to select a power source for the PULSE probe there were 12 factors which I took into consideration: 1)Duration 2)Mission 3)Availability 4)Reliability 5)Weight 6)Compatibility 7)Environment 8)Power level 9)Area 10)Cost 11)Volume 12)Hazard. Since the mission duration of our probe is about 16 years the selection of power source was limited to nuclear power, either from decay of an isotope or a nuclear reactor. Batteries were also considered for storing the electrical energy provided by the power source. The approach taken consisted of listing the 12 factors and rating the sources from 1 to 10(highest) on the quality of performance related to each of the 12 factors as shown in figure 6.5.

The results from this trade study eliminated the nuclear reactor as a power source but showed that batteries should be further considered as energy storage devices for the RTGs. But when looking at the predicted power to weight ratio of both the RTG(12 W/kg) and the Ni-Cd battery (10 W-Hr/kg) in the year 2000 the choice was that the RTGs were the only power source that was going to be used on the PULSE probe (Ref. 10, pp.1-45).

The next step in developing the power system was finding out how much power the power system would have to put out at peak operating loads. Figure 6.6 shows a list of the subsystems and the power that each subsystem requires at peak level. Figure 6.7 shows the percentage of power each subsystem requires of the total power. A total power system requirement of 372.94 W is needed upon arrival at Pluto.

The isotope selected for this mission is Pu 238, with a half life of 87 years. This isotope has been proven by earlier space missions and often exceeded its original design life requirements. Some studies have used a design lifetime of 10 years for the RTG and found that the RTG has a 20% reduction in power at the end of the projected 10 year life (Ref. 10, pp.1-48).

The PULSE probe's RTGs will have to supply power for at least 16 years. This results in a 70% reduction in 16 years which shows that at launch the PULSE probe will have 529.7 W of power that would diminish to the amount needed at Pluto (See appendix for these calculations). No safety margin is needed with these figures because the Pu 238 RTG "has operated considerably longer than their original design life requirements" (Ref. 10, pp.1-44). From the total power needed at launch a calculation was made to determine the mass of RTG needed. The mass of RTG needed is 44.40 kg, which would require 23 slices of fuel cells in the Modular Isotopic Thermoelectric Generator (Ref. 12, pp.340) (See appendix for calculations). The RTG fuel capsule is designed to withstand intact reentry should there be a mission failure or abort.

The electrical power from the RTG will go to the Power Conditioning Unit which will regulate the voltage and convert the DC power into whatever form it needs to be in for the applied loads. This will depend upon the voltages needed by the instruments and if they are powered by AC or DC voltage (Figure 6.8).

Fig.6.6

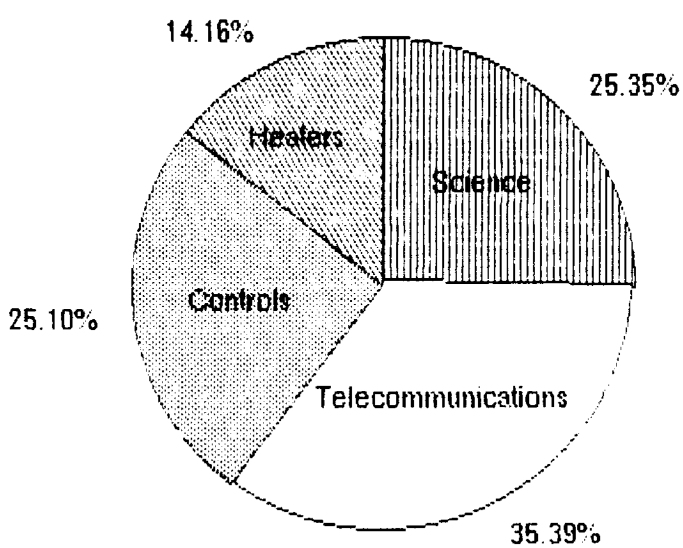
Power Supply Determination

	Reactor	RTG	Battery
Duration	6	8	4
Mission	2	10	6
Availability	6	8	8
Reliability	6	10	10
Weight	4	8	8
Compatibility	6	10	8
Environment	8	8	8
Power level	10	8	6
Area	4	8	6
Cost	4	6	10
Volume	2	8	8
Hazard	6	8	10
Total	64	100	92

Fig.6.7 **Power Systems**

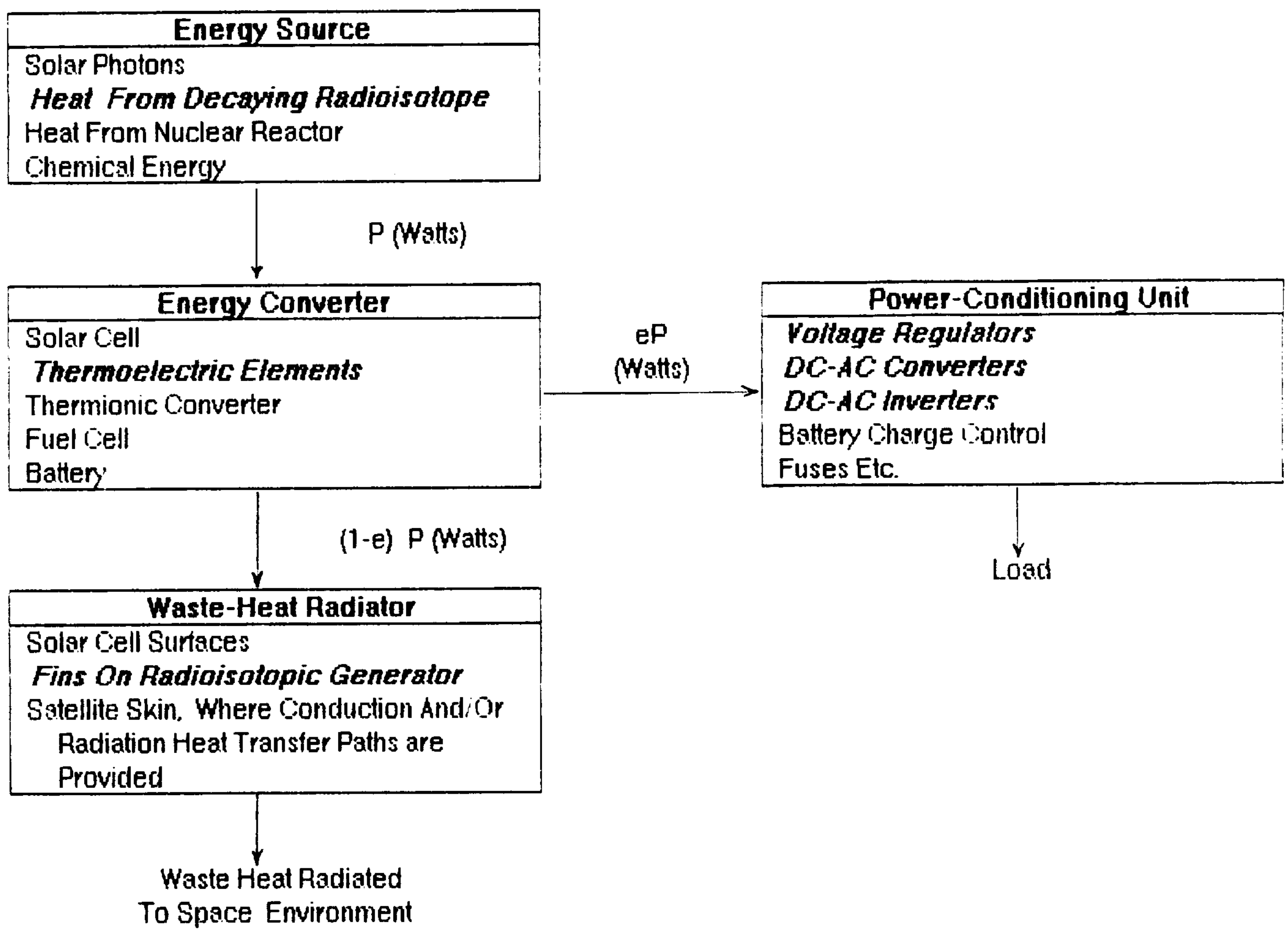
System function	Power required at peak levels (Watts)
Science	78.78
Telecommunications	110
Control	5
Receiver	10
Amplifier	70
Data samplin, encoding, and decoding	20
Data storage	5
Spacecraft control	78
Sequencing and command	10
Sun sensors	3
Canopus tracker	10
Gyros	15
Electronics	40
Heaters	44
Total system requirements	310.78
Conversion loss (20%)	62.16
Total power requirement	372.94

Fig.6.8 **Power Subsystems**



Power System

Fig.6.9



Appendix I

Propulsion

Isp = specific impulse
V = delta V needed
ε = structural coefficient
ML = mass of the payload (spacecraft)
Thrust = thrust given by each of the stages
c = exhaust velocities
f(α) = function used for Newton's Approximation
fprime(α) = derivative of f(α)
α = Lagrange multiplier
R = mass ratio
MSP = mass of structure and propellant of that stage
M = mass of that stage plus payload weight
Ms = mass of the structure of that stage
Mp = mass of the propellant on that stage
Mo = total mass of the launch vehicle and spacecraft
λ = payload ratio
Massflow = massflow of that stage
Burntime = burntime of that stage

Base units:

$sec \equiv 1T$ $\quad kg \equiv 1M$ $\quad m \equiv 1l$

Normal units:

$N \equiv kg \cdot \frac{m}{sec^2}$ $\quad km \equiv 1000 \cdot m$ $\quad lb \equiv \frac{kg}{2.2}$ $\quad lbf \equiv 4.4 \cdot N$

Constants:

This shows only one launch vehicle. This process was done 3 times
A chart with all the values is in the text

$Isp_1 := 310 \cdot sec$ $\quad Isp_2 := 320 \cdot sec$ $\quad Isp_3 := 360 \cdot sec$

$g := 9.8 \cdot \frac{m}{sec^2}$ $\quad V := 8.974 \cdot \frac{km}{sec}$ $\quad ML := 670.40 \cdot kg$

Assuming structural coefficients to be the same for Titan and Ariane
(Actual Ariane values)

$\varepsilon_1 := .0696$ $\quad \varepsilon_2 := .0957$ $\quad \varepsilon_3 := .1008$

$thrust_1 := 899000 \cdot lbf$ $\quad thrust_2 := 177000 \cdot lbf$ $\quad thrust_3 := 14000 \cdot lbf$

Equations:

$i := 1 .. 3$

$c_i := Isp_i \, g$

Iteration using Newton's approximation

$$f(\alpha) := V - \sum_i c_i \ln\left[\frac{\alpha\, c_i - 1 \frac{km}{sec}}{\alpha\, c_i\, \varepsilon_i}\right]$$

c_i
$3.038 \cdot 10^3$ length^1 time^{-1}
$3.136 \cdot 10^3$ length^1 time^{-1}
$3.528 \cdot 10^3$ length^1 time^{-1}

$$fprime(\alpha) := \sum_i \left[c_i \left[\left[\frac{c_i}{\alpha\, c_i - 1 \frac{km}{sec}}\right] - \frac{1}{\alpha \cdot .99999999999}\right]\right] \cdot -1$$

$j := 0 .. 20$ $\qquad x_0 := .43$

$$x_{j+1} := until\left[\left|f\left[x_j\right]\right| - .0001 \frac{km}{sec}, x_j - \left[\frac{f\left[x_j\right]}{fprime\left[x_j\right]}\right]\right]$$

$n := size(x)$ $\qquad \alpha := x_n$ $\qquad \alpha = 0.4$

$$R1 := \frac{\alpha\, c_1 - 1 \frac{km}{sec}}{\alpha\, c_1\, \varepsilon_1} \qquad R2 := \frac{\alpha\, c_2 - 1 \frac{km}{sec}}{\alpha\, c_2\, \varepsilon_2} \qquad R3 := \frac{\alpha\, c_3 - 1 \frac{km}{sec}}{\alpha\, c_3\, \varepsilon_3}$$

$R1 = 2.557$ $\qquad R2 = 2.128$ $\qquad R3 = 2.898$

$$MSP3 := \frac{ML - R3 \cdot ML}{R3 \cdot \varepsilon_3 - 1} \qquad MSP3 = 1.798 \cdot 10^{3}$$

$$MSP2 := \frac{MSP3 + ML - R2 \cdot MSP3 - R2 \cdot ML}{R2 \cdot \varepsilon_2 - 1} \qquad MSP2 = 3.495 \cdot 10^{3}$$

$$MSP1 := \frac{MSP2 + MSP3 + ML - R1 \cdot MSP2 - R1 \cdot MSP3 - R1 \cdot ML}{R1 \cdot \varepsilon_1 - 1} \qquad MSP1 = 1.129 \cdot 10^{4}$$

$M_{03} := MSP3 + ML$ $\qquad M_{02} := MSP3 + MSP2 + ML$ $\qquad M_{01} := MSP3 + MSP2 + MSP1 + M$

$M_{03} = 2.468 \cdot 10^{3}$ mass $\qquad M_{02} = 5.963 \cdot 10^{3}$ mass $\qquad M_{01} = 1.726 \cdot 10^{4}$ mass

$Ms_1 := \varepsilon_1 \cdot MSP1$ $\qquad Ms_2 := \varepsilon_2 \cdot MSP2$ $\qquad Ms_3 := \varepsilon_1 \cdot MSP3$

$Ms_1 = 786.017$ mass $\qquad Ms_2 = 334.515$ mass $\qquad Ms_3 = 125.111$ mass

$Mp_1 := MSP1 - Ms_1$ $\qquad Mp_2 := MSP2 - Ms_2$ $\qquad Mp_3 := MSP3 - Ms_3$

$Mp_1 = 1.051 \cdot 10^{4}$ mass $\qquad Mp_2 = 3.161 \cdot 10^{3}$ mass $\qquad Mp_3 = 1.672 \cdot 10^{3}$ mass

$M_0 := MSP1 + MSP2 + MSP3 + ML$ $\qquad M_0 = 1.726 \cdot 10^{4}$ mass

$$\lambda_1 := \frac{M_{02}}{M_0 - M_{02}} \qquad \lambda_2 := \frac{M_{03}}{M_{02} - M_{03}} \qquad \lambda_3 := \frac{ML}{M_{03} - ML}$$

$$\lambda_1 = 0.528 \qquad \lambda_2 = 0.706 \qquad \lambda_3 = 0.373$$

$$\mathrm{Massflow}_i := \frac{\mathrm{thrust}_{i,1}}{c_i} \cdot \frac{1}{1000} \qquad \mathrm{Burntime}_i := \frac{\mathrm{Mp}_i}{\mathrm{Massflow}_i}$$

$\mathrm{Massflow}_i$
$1.302 \cdot \mathrm{mass}^{1} \cdot \mathrm{time}^{-1}$
$0.248 \cdot \mathrm{mass}^{1} \cdot \mathrm{time}^{-1}$
$0.017 \cdot \mathrm{mass}^{1} \cdot \mathrm{time}^{-1}$

$\mathrm{Burntime}_i$
$8.07 \cdot 10^{3} \cdot \mathrm{time}^{1}$
$1.273 \cdot 10^{4} \cdot \mathrm{time}^{1}$
$9.579 \cdot 10^{4} \cdot \mathrm{time}^{1}$

FRANCE

ESA/Arianespace

Ariane 2	CNES/Arianespace	1	4 x Viking 5 liquid	Aerospatiale/SEP	L-140	N_2O_4/UH25	601,000	12.5	59.8	490,000	4,795	3,200
		2	1 x Viking 4 liquid	ERNO/SEP	L-33	N_2O_4/UH25	177,600	8.5	37.6	total	geostationary	
		3	1 x HM-7B liquid	Aerospatiale/SEP	H-10	LOX/LH_2	14,000	8.5	34.2		transfer	
Ariane 3	CNES/Arianespace	1	4 x Viking 5 liquid	Aerospatiale/SEP	L-140	N_2O_4/UH25	601,000	12.5	59.8	530,000	5,890	3,750
		1	2 x P7.3 solid	BPD	PAP	Solid	250,000	3.5	26.2	total	geostationary	
		2	1 x Viking 4 liquid	ERNO/SEP	L-33	N_2O_4/UH25	177,500	8.5	37.6		transfer	
		3	1 x HM-7B liquid	Aerospatiale/SEP	H-10	LOX/LH_2	14,000	8.5	34.2			
Ariane 4[a]	CNES/Arianespace Arianespace	1	4 x Viking 5 liquid	Aerospatiale/SEP	L-220	N_2O_4/UH25	601,000	12.5	82.5	525,000	4,[illegible]	—
		1[a]	2-4 x Viking 6 liquid	ERNO/SEP	L-36	N_2O_4/UH25	152,000	7.1	62.3	to	to	
		1[a]	2-4 x P9.5 solid	BPD	P9.5	Solid	146,000	3.5	39.2	1,033,000	8,250	
		2	1 x Viking 4 liquid	ERNO/SEP	L-34	N_2O_4/UH25	177,000	8.5	37.6	(AR44L)	geostationary	
		3	1 x HM-7B liquid	Aerospatiale/SEP	H-10	LOX/LH_2	14,000	8.5	34.2		transfer, [illegible]	

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Power

20% decrease in power over 10 years (Ref. 10, pp.1-48)

$N(t)$ = percentage of power after t years

N_o = percentage of power at launch

k = decay constant

t = time

$$N(t) = No\ e^{-kt}$$

$$.80 = 1\ e^{-k(10)}$$

$$k = -\ln(.80)/10$$

$$k = 0.022314$$

$$N(t) = 1\ e^{-(0.022314)(16.005)}$$

$$N(t) = 0.69967$$

This is a 30% decrease over 16 years

Total power needed/70% = Power at launch/100%

372.94/70% = Power at launch/100%

Power at Launch = 529.69 W

Assuming (12W/kg) power to weight ratio predicted for the year 2000 (Ref. 10, pp.1-45)

529.69 W/12W/kg= 44.40 kg of RTG at launch

MITG Generator give 23.5W/slice (Ref. 12, pp.340)

529.69 W / 23.5W/slice = 22.54 slices approximately 23 slices

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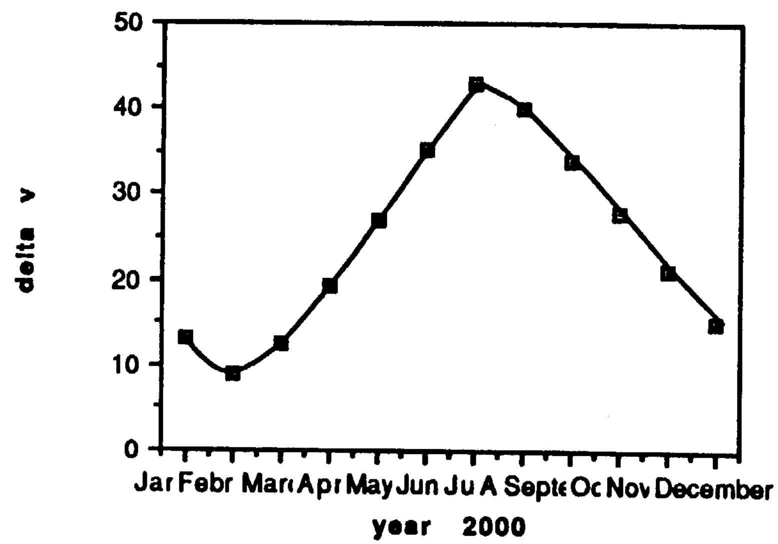
The spacecraft PULSE uses much off-the-shelf hardware from Voyager and other planned probes. New technology is only applied if it would include a more reliable and less costly trade-offs, as in the case of onboard computers. PULSE willyield quality science at low cost by using incorporation of off-the-shelf products, choosing radiation-hardened version of widely available microprocessor and integrated-circuit chips supported by efficient software. In general, proven techniques were used throughout the entire design.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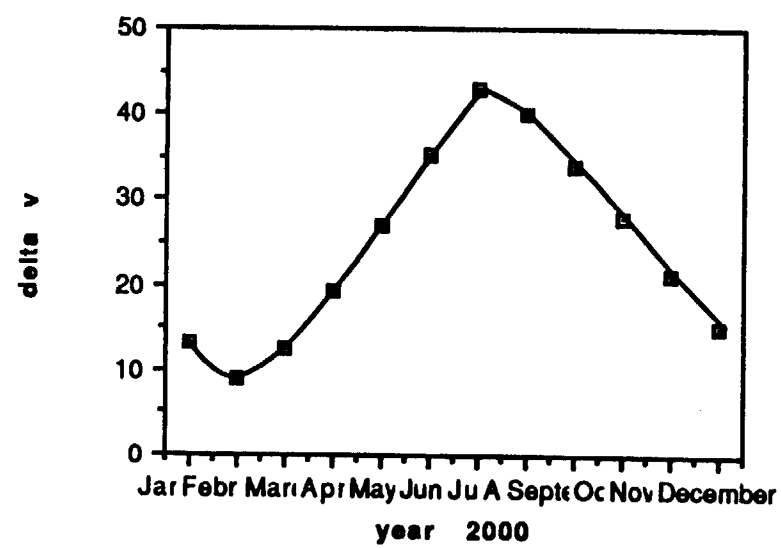

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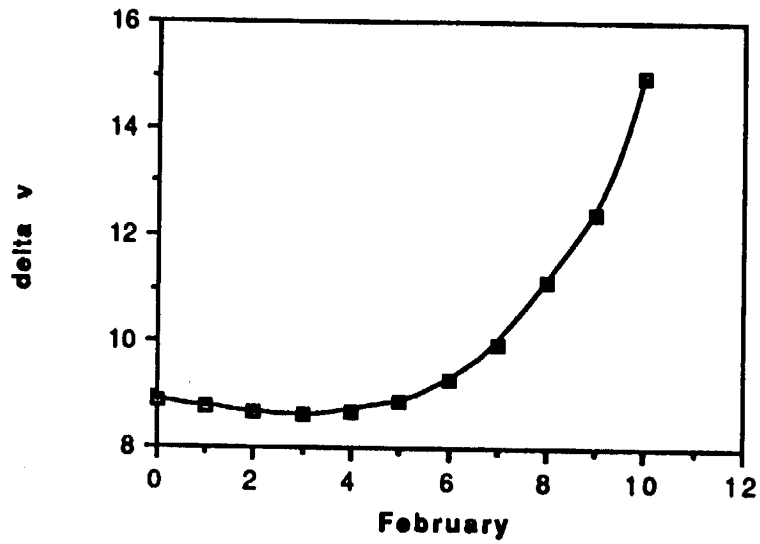

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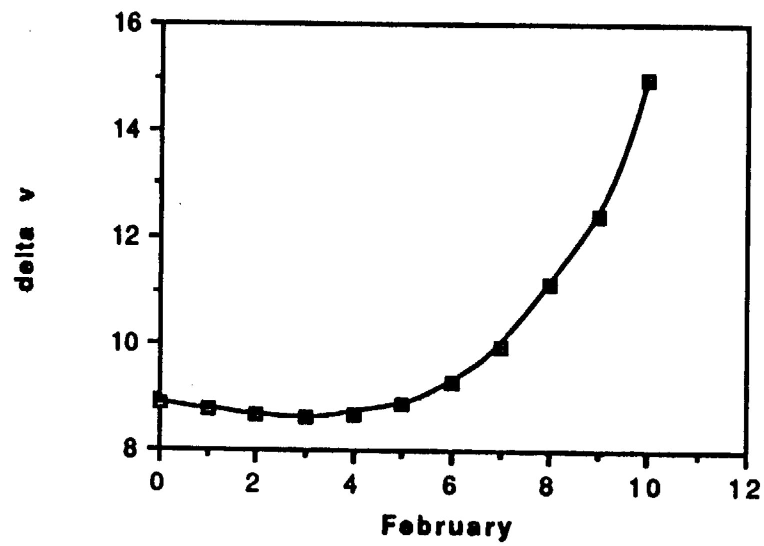
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Costing for PULSE

Category	Cost (FY 88 Dollars)
Structure	59,988,162.98
Thermal Control	11,037,938.33
Propulsion	412,927,670.50
Attitude & Articulation	62,614,609.37
Telecommunications	64,098,191.33
Antennas	13,043,018.66
Command & Data Handling	24,500,108.53
RTG Power	37,386,446.55
Line-Scan Imaging	170,454,335.10
Particle & Field Instruments	71,222,537.72
Remote Sensing Instruments	29,154,302.64
System Support & Ground Equipment	280,062,535.20
Launch + 30 Days Ops & Ground S/W	57,185,698.78
Image Data Development	6,957,007.47
Science Data Development	11,487,733.40
Program Management	17,365,267.83
Flight Operations	258,722,216.60
Data Analysis	115,984,760.70
TOTAL	**1,704,192,542.00**

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SUBSYSTEM SUMMATION

MISSION CLASS: FLYBY

TRAJECTORY: DIRECT PATH FROM EARTH TO PLUTO

DELTA V REQUIRED: 8.606 KM/SEC (FROM PARKING ORBIT)

LAUNCH DATE: JANUARY 30, 2003

ARRIVAL DATE AT PLUTO: FEBRUARY 1, 2019

MISSION DURATION: 16.005 YEARS

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Table 3.2

Weights and Power for PEP Instrumentation

Instrument	Power (W)	Mass (kg)
ISS	20	28*
MAG	2.2	5.6
NIMS	13	18
PPR	4.5	4.8
VVS	5.33	4
PLS	10*	12
EDP	10*	9
PWS	8.4*	6
CRS	5.35	7.5

* Values are estimates

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D.D. Group 5

P-97

N91-16052

The Copernicus Project

Proposal for AAE 241
University of Illinios
April 24, 1990

Group Six:
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Hans Polte
Paul Kepes
Kevin Walker
Jeff Jacobs
Stephen Williams

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Kevin J. Walker 357-48-7444

Stephen Williams: Structures

Stephen Williams 226-27-9359

Executive Summary

The Copernicus spacecraft, to be launched on May 4, 2009, is designed for scientific exploration of the planet Pluto. The main objectives of this exploration is to accurately determine the mass, density, and composition of the two bodies in the Pluto-Charon system. A further goal of the exploration is to obtain precise images of the system.

The spacecraft will be designed for three axis stability control. It will use the latest technological advances to optimize the performance, reliability, and cost of the spacecraft. Due to the long duration of the mission, nominally 12.6 years, the spacecraft will be powered by a long lasting radioactive power source. Although this type of power may have some environmental drawbacks, currently it is the only available source that is suitable for this mission.

The planned trajectory provides flybys of Jupiter and Saturn. These flybys provide an opportunity for scientific study of these planets in addition to Pluto. The information obtained on these flybys will suppliment the data obtained by the Voyager and Galileo missions.

Table of Contents

ACRONYMS

AACS: Attitude, Articulation, and Control System

ASTROS: Advanced Star/Target Reference Optical System

CCC: Command, Control, and Communication

CCD: Charge-Coupled Device

COM: Center of Mass

FDC: Flight Data Subsystem

FORS: Fiber Optic Rotation System

HECP: High Energy Charged Particle

HDA: Harmonic Drive Actuator

ISS: Imaging Science Subsystem

IIS: Infrared Interferometer Spectrometer

kbps: Kilobits Per Second

LECP: Low Energy Charged Particle

MITG: Modular Isotopic Thermoelectric Generator

MMPC: Mission Management, Planning and Costing

NAC: Narrow Angle Camera

PP: Plasma Particle

ACRONYMS (cont.)

RCS: Reaction Control System

RTG: Radioisotope Thermoelectric Generator

TCM: Trajectory Correction Maneuver

S/C: Spacecraft

UVS: Ultraviolet Spectrometer

ΔV: Delta Velocity

WAC: Wide Angle Camera

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INTRODUCTION

Introduction

The Copernicus Project proposal describes a Phase A design for an unmanned mission to Plutoian space for the purpose of scientific inquiry. This paper proposes that the spacecraft be designed, built, and launched in an effort to increase our knowledge of the outer Solar System and, in particular, the Pluto-Charon system. Thus far Pluto is the only planet that has not been visited and investigated by a space probe.

In order to insure an efficient and successful spacecraft and to bring focus to the overall mission, the Copernicus Project proposal will adhere to various mission guidelines and design requirements. The following is a list of the spacecraft primary design requirements.

- The spacecraft must be unmanned.
- The spacecraft must be launched in the first decade of the twenty-first century.
- The spacecraft should be reliable and easy to operate.
- The spacecraft should use off the shelf hardware whenever possible.
- The spacecraft should not use materials or techniques expected to be available after 1999.
- On-orbit assembly should be identified and minimized.
- The launch vehicle to be used must be identified and the interfaces must be compatible.
- The design must be flexible enough to perform several possible missions.
- The design lifetime must be sufficient to carry out the mission plus a reasonable safety margin.
- The spacecraft must use the latest advances in artificial intelligence.
- The design will stress reliability, simplicity, and low cost.
- Four spacecraft will be built.
- Give an implementation plan for production of a final product.

In an effort to adhere to these design requirements and to create an original and unique proposal, the project is divided into six subsystems. Each subsystem is responsible for the design of a specific area of the mission and the identification of any interactions between the subsystems. An additional responsibility of each subsystem is to optimize the performance, weight, and cost of the individual subsystem in order to optimize those parameters for the overall mission design. A list of the subsystems and their major responsibilities follows.

Structures: Responsible for material selection for major spacecraft components, component placement, thermal control for the spacecraft, calculation of spacecraft inertia and center of mass, and production planning.

Mission Management, Planning and Costing: Responsible for mission type selection, trajectory planning, launch vehicle selection, mission timeline planning, and mission costing.

Command, Control, and Communication: Responsible for the quality of the spacecraft computers, the information storage capability of the spacecraft, and insuring that the communication link with the spacecraft is available at all times.

Power and Propulsion: Responsible for providing adequate power supplies to the spacecraft components during all mission phases, propellent selection, and propulsion unit selection and sizing.

Science Instrumentation: Responsible for planning the mission science objectives, planning the mission science timeline, and scientific instrument selection.

Attitude and Articulation Control: Responsible for attitude control of the spacecraft, maintaining antenna pointing requirements, trajectory correction maneuvers, science maneuvers, and stability throughout the mission.

STRUCTURES

Structure Subsystem: Introduction

The responsibility of the structure subsystem for the Pluto project is to stress reliability, simplicity, and low cost in the areas of material selection, thermal control, and overall spacecraft design. Subjects to consider in fulfilling this responsibility are minimizing the spacecraft weight, minimizing the amount of on-orbit assembly of the spacecraft, and insuring a design lifetime sufficient to carry out the mission plus a safety margin. An additional responsibility is to provide an implementation plan for production of the final product. To meet these requirements the structure subsystem is divided into the following areas of consideration:

1. Drawings of the spacecraft
2. Placement of the spacecraft components to meet requirements
3. Mass and inertia of the spacecraft
4. Material selection
5. Thermal control
6. Launch vehicle compatibility
7. On-orbit assembly
8. Production of the final product
9. Interactions with other subsystems

Drawings of the Spacecraft

Drawings of the spacecraft are provided to enhance the reader's conception of the component placement and the overall spacecraft design. The major spacecraft components included in the drawings are the bus, propellent tank, main propulsive unit, three boom extensions, RTG, scan platform, and antenna unit. Major spacecraft dimensions are provided in meters. Two views of the spacecraft will provide the reader with a clear idea of the spacecraft configuration.

The spacecraft axis was selected such that the origin coincides with the geometric center of the bus. The Z-axis points out along the antenna mast, the X-axis points out along the magnetometer boom, and the Y-axis points out along the science boom to form a standard righthanded coordinate system.

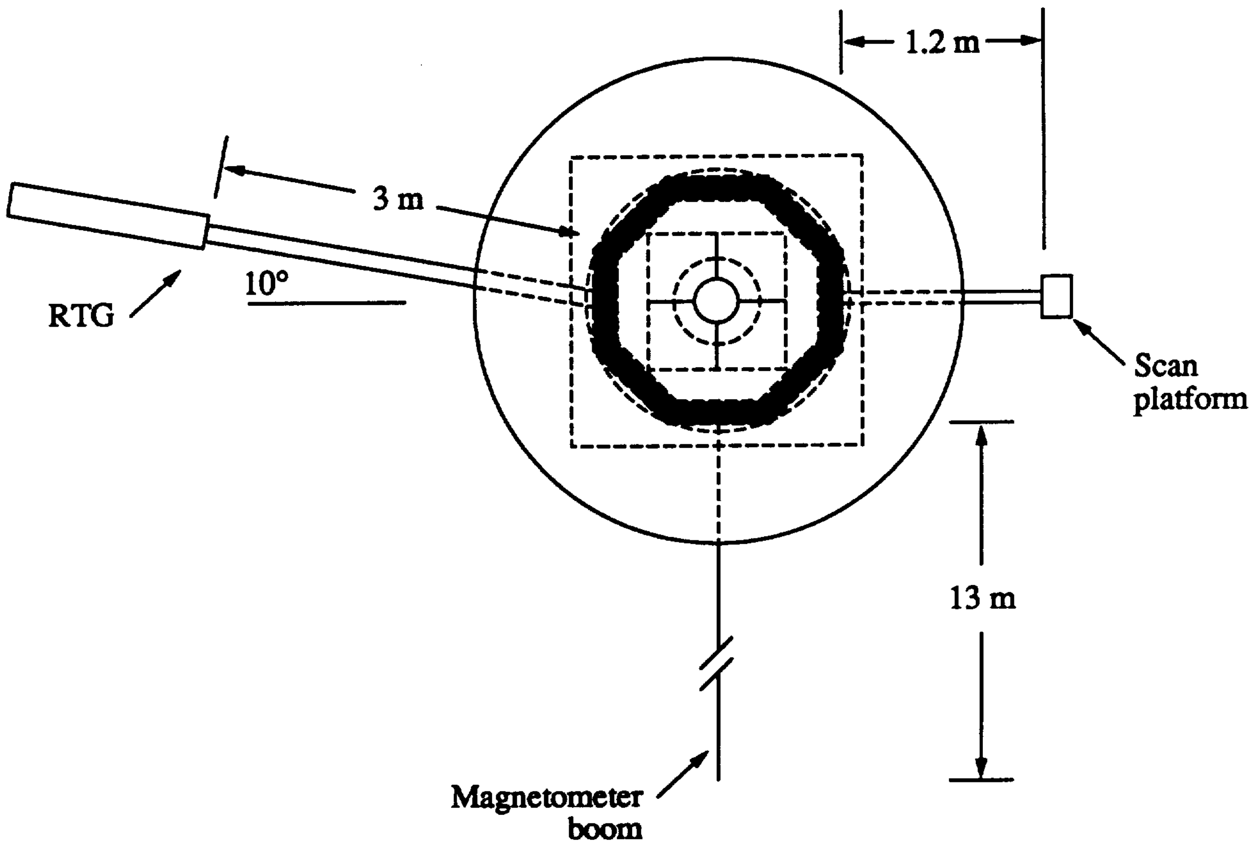

Figure 1A

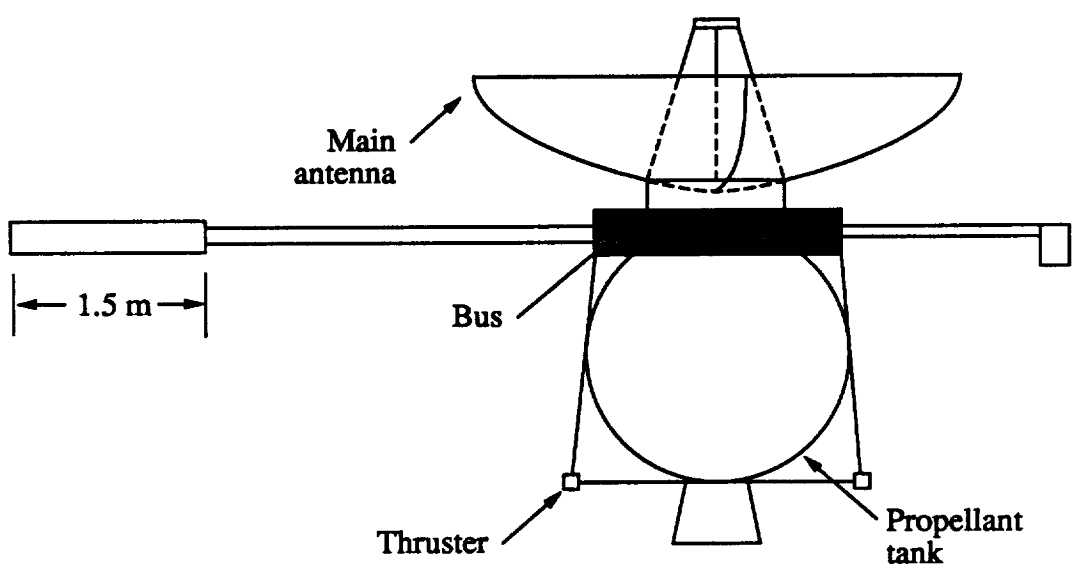

Figure 1B

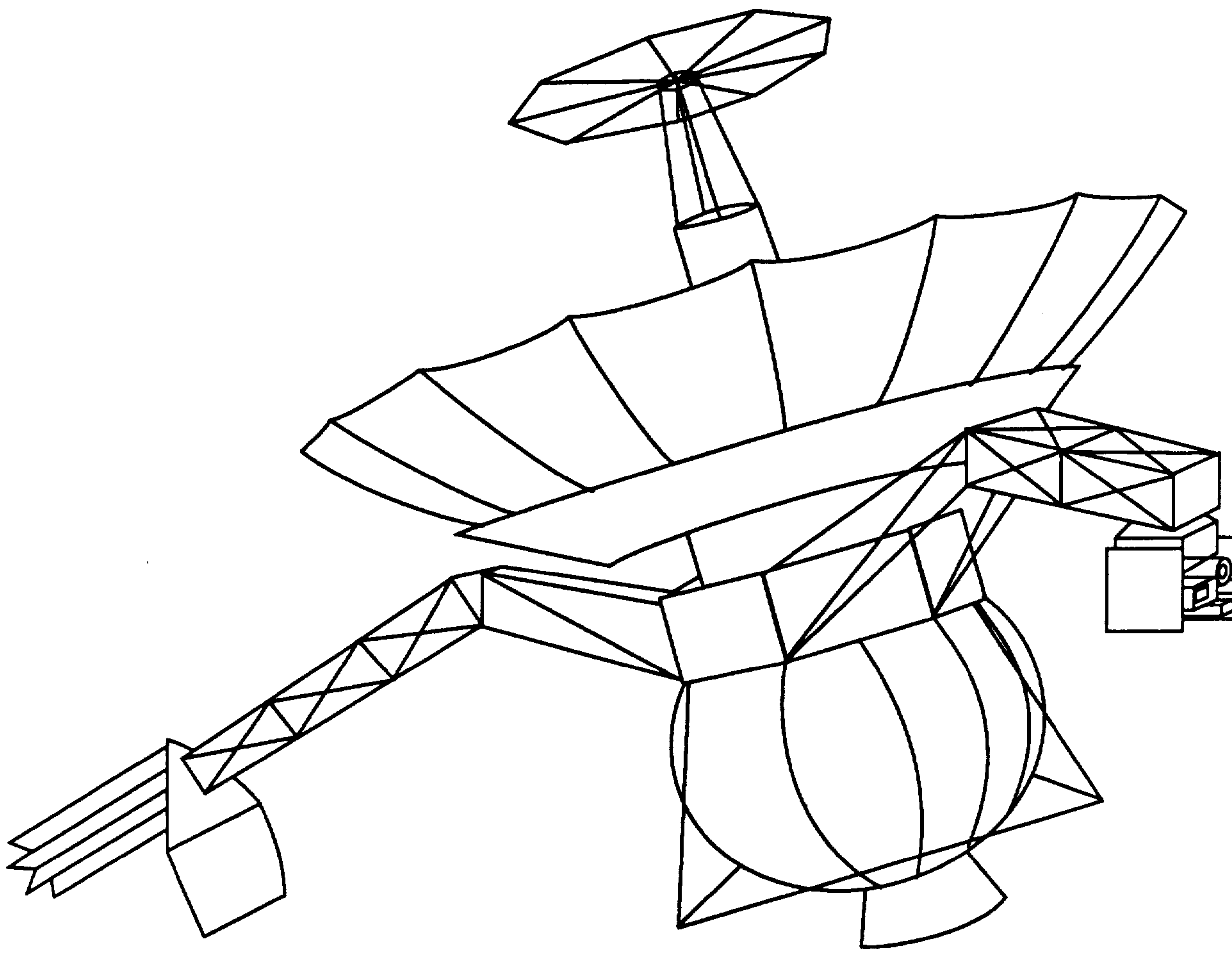

Figure 1C
Copernicus

Placement of Spacecraft Components to Meet Requirements

The driving requirements of this mission are reliability, simplicity, and low cost. From these primary requirements come several derived requirements that influence the positioning of the individual spacecraft components. These derived requirements are: radiation protection for all spacecraft components, the scientific instruments must have a clear field of view, no component that would disrupt communications should be placed near the antenna, the main propulsive unit should create a line of force through the spacecraft center of mass (COM), the components in the bus must be compact to aid in thermal control of the bus, and the magnetometer must be isolated from the interference of other spacecraft components.

The radioisotope thermoelectric generator (RTG) emits radiation that is damaging to other spacecraft components. To minimize this radiation damage the RTG should be placed as far as possible from all other spacecraft components. The distance that the RTG can be placed from the main spacecraft structure is limited by the strength of the RTG boom and spacecraft COM considerations. In an effort to keep the COM on the Z-axis for attitude control simplification, the RTG will be placed approximately 3 meters from the bus and at an angle of 10^{o} off the negative Y-axis in the XY plane. For additional radiation protection, a metal shield will be placed at the end of the RTG boom between the RTG and the main body of the spacecraft.

The placement of the scan platform must provide an adequate viewing range for the scientific instruments. This is one of the most important placement requirements. If this requirement is not met, then the success of the mission will be limited. The scan platform will be placed on a 1.2m boom that extends 0.3m beyond the rim of the antenna. This placement was achieved by a tradeoff of field of view and the previously mentioned COM restriction. Also, the scan platform will be placed such that the spacecraft main body is between the platform and the RTG for redundant radiation protection.

Communication is also essential for the success of the mission. In an effort to increase reliability, any components that are placed within the antenna's field of transmission or reception should be transparent to the antenna. A better placement technique is to leave this area of the antenna free of any components at all. The second technique is simpler than the use of antenna transparent components and it was therefore selected.

To prevent any unwanted torques while the main propulsive unit is in operation, the unit will be oriented so that its line of force coincides with the Z-axis of the spacecraft. As previously stated, all spacecraft components will be positioned so that the spacecraft COM lies on the Z-axis.

The components housed within the bus will be placed in a compact manner. This technique reduces the overall volume of the bus and therefore the volume that requires the most thermal control. The method in which this reduction in thermal control cost is achieved will be discussed in a later section. The compact placement of the components within the bus helps to reduce the mission cost and thereby helps to fulfill a primary mission requirement.

A final placement requirement involves the magnetometer. The magnetometer must be placed as far as possible from the other spacecraft components to reduce the amount of interference encountered from the other components. Again, the distance that the magnetometer can be placed from the main spacecraft assembly is restricted by COM placement, the strength of the magnetometer boom, and the cost per unit length of the boom.

Mass and Inertia of the Spacecraft

Mass estimates are provided only for the major components of the spacecraft. The following mass estimates are derived from other subsystem requirements, considerations, and calculations.

Table 1-A. Component Masses

Component:	Mass (kg):
Antenna	5
Antenna Base	45
Bus (includes Structure, Thermal Control, and Cabling)	270
Computers	100
Science Platform	111
Science Boom	35
Magnetometer Boom	5
RTG Boom	5
RTG	60
Propulsion Unit Tank	120
Propellent	1500-2000

Total spacecraft mass (unfuelled): 756 kg

The inertia of the spacecraft is calculated with the aid of a computer program. The inertia and COM of individual components are calculated by hand and these results are input into the program which calculates the overall spacecraft inertia and COM. The individual components are idealized into geometric shapes to simplify the inertia calculations as described in the structure section appendix.

In an effort to simplify the placement of the attitude thrusters and the main propulsive unit, the spacecraft COM should lie on the Z-axis and as close to the geometric center of the bus as possible. Several trials were performed in which the lengths of the science boom and the RTG boom were varied. An additional variable was the angle between the RTG boom and the negative Y-axis in the XY plane. On the ninth trial the spacecraft COM was within approximately 0.5 cm of the Z-axis and approximately 11 cm below the geometric center of the bus. This result was obtained with the unfuelled configuration. This position of the COM is adequate for the purposes of this preliminary design report.

The inertia and COM for the unfuelled configuration of the ninth trial is:

Table 1-B. Inertia of Copernicus

Body Name: Copernicus			
Inertia Matrix:	2334.0560	-155.8637	-.3384
	-155.8637	700.0375	-. 4187
	-.3384	-.4187	2724.9290
Body COM:	.0039	.0048	-.1147
Body Mass:	756.0000		
Number of Bodies:	9		
Principal Inertia Matrix:			
	2348.7910	.0000	.0000
	.0000	685.3029	.0000
	.0000	.0000	2724.9290
Eigenvector Matrix:			
	.9956	.0941	- .0008
	-.0941	.9956	- .0001
	.0008	.0002	1.0000

Material Selection

There are several factors to consider in the material selection process. First, to comply with the primary mission requirements, the materials should be light weight, low cost, and should reliably fulfill their design function. Additional material selection considerations include: radiation damage threshold, contamination resistance, thermal characteristics, strength, stiffness, and general structural qualities. These characteristics must be carefully considered when selecting materials for the spacecraft.

The main purpose of this mission is scientific exploration of Plutoian space. Therefore it is essential that the science instruments be kept operational. Contaminants such as atomic oxygen, outgassed materials, and cosmic debris will accumulate on instrument surfaces over time and impede their performance. Since the mission is of such long duration, contamination protection of the instruments must be a major factor in material selection.

One method of protecting the instruments is by installing a permanent cover which is transparent to the instrument. A second means of protection is the retractable cover design. This design involves moving parts and should be used only where absolutely necessary in an effort to enhance simplicity. If the retractable cover should fail to open, then the success of the mission would be limited.

A redundant method of radiation protection is achieved by placing a metal shield between the RTG and the main spacecraft body. An aluminum shield was selected due to its low cost, light weight, and high radiation damage threshold. Composites should not be used for this application due to their susceptibility to radiation damage.[1]

An application that is well suited for composite materials is the main antenna. The composite can be easily molded into the unique antenna shape. Also, because of their low coefficient of thermal expansion and high thermal conductivity, composites can be used in systems which require high thermostructural stability like the antenna dish.[2]

For the main structural supports of the spacecraft, titanium should be used where strength and thermal stability is important. Graphite epoxy can be used in secondary truss supports and stiffeners. Aluminum is attractive for its strength to weight ratio, availability, low cost, and because it is space proven.

In situations where the stiffness of a structural member is crucial, beryllium will be used instead of titanium. The modulus of elasticity of beryllium is 2.5 times that of titanium and beryllium is considerably lighter in weight. Although beryllium is more costly to produce than titanium, beryllium's weight savings makes it less costly than titanium to put into orbit.[3]

The use of cosmic ray resistant parts for the computer's electronic components will depend on their performance on the Galileo probe.

Sandia National Laboratories developed these components in an effort to reduce the number of single event upsets in the computer's logic and memory.[4] If these components prove successful in reducing the number of computer sequence failures and if the cost is reasonable, then cosmic ray resistant parts should be incorporated into the Pluto probe's computer for enhanced reliability and performance.

Thermal Control

Thermal control will insure that each part of the spacecraft will have an appropriate thermal environment for operation. The different components will require significantly different thermal environments so that temperature gradients will be present throughout the spacecraft. Thermal control will be further complicated by the changing thermal surroundings as the mission progresses. The three most significant phases are: thermal control on Earth and during launch, thermal control in space close to the sun (0.5-3 AU), and thermal control in the outer solar system.

The problem of thermal control is best solved by examining the major components of the spacecraft.

Bus: The major considerations for thermal control of the bus are isolation from solar heating, internal coupling to prevent temperature gradients, and heat rejection at external bus surfaces.[5] A very cost and weight efficient method of preventing solar heating in the bus is by the use of multilayer insulation blankets. This passive thermal control technique makes use of the unique insulation properties of multilayer designs. Redundancy is also achieved by using multiple layers. The material is selected for minimum heat transmission except for a few layers of very tough material such as Teflon for micrometeoroid protection. The internal coupling is achieved by positioning the internal components as compactly as possible. This technique produces a smaller volume to be thermally controlled and thus the cost of thermal control is reduced. This helps meet the low cost mission requirement. The heat rejection phase is accomplished by transporting waste heat from the interior of the bus to the external bus surfaces via a system of thermal switches. At points along the external bus surface are heat radiators in the form of thermostatically controlled louvers. There will be several of these louver sites for redundancy.

RTG: The RTG produces large amounts of heat to be converted into electrical power for the spacecraft. Due to radiation protection considerations, the RTG is relatively isolated from all other spacecraft components. This isolation also serves as an excellent thermal barrier between the RTG and the spacecraft. Any waste heat produced by the RTG can easily be rejected into space by an array of metal fins that act as passive heat radiators.

Thrusters: The hydrazine thrusters will be thermally controlled by strip heaters constructed of printed heating element circuits imbedded in Kapton film.[6] These heaters will be placed on the catalyst bed of the thrusters to produce temperatures well above 500K. The hydrazine fuel lines will be heated by wrapping wire heating elements around the fuel line.

Science Instruments: The great design flexibility of the printed circuit strip heaters mentioned above will allow them to provide thermal control to the science instruments as well as the thrusters. The design temperature for the science instruments is approximately 140K which is well within the thermal range of the strip heaters. To help meet the requirement of redundancy in all spacecraft systems, two strip heaters will be provided for every science instrument and every thruster. This increase in thermal control should not produce a drastic increase in overall spacecraft weight due to the very small mass of the strip heaters.

Launch Vehicle Compatibility

The spacecraft must be compatible with the selected launch vehicle. This means that all interfaces between the spacecraft and the launch vehicle must be selected for compatibility. Also, the dimensions of the spacecraft cannot exceed the payload dimensions of the chosen launch vehicle.

The launch configuration is approximately cylindrical in shape. The approximate dimensions of this cylinder are: width=3.7 m and length = 4.5 m. The width c- ponds to the antenna diameter. The antenna is a one pie- and is similar in design to the antenna used on th t. The fact that the antenna is not foldab' unch vehicle selection at all. The leng ate any problems with the payload a

what about interface(s)

The launch vehicle must not create any e interfaces include thermal contr assistance in thermal control while nch pad and while the launch ve ther interfaces may include power interfaces, and the mountings that h position inside the launch vehicle. All these be compatible.

On-Orbit Assembly

The Copernicus will be a complete unit in its launch configuration. No assembly will be required while in orbit. However, there will be several boom deployments and general transformations of the spacecraft from its launch configuration to its cruise configuration while in orbit. Separation from the launch vehicle and upper stage will be achieved by pyrotechnic methods such as explosive bolts.

These deployments will be made while the spacecraft is in LEO. This will enable a repair and/or rescue attempt in the event of a deployment failure. If the deployments are made in GEO or on route to Pluto and a deployment failure occurs, then repair attempts would be much more difficult to engineer. Deployment of the booms in LEO will help improve mission reliability.

Production of the Final Product

The production of the Pluto probe will be a multistep process of design, parts construction, system integration, and possible redesign. In each of these phases testing for quality and reliability is essential. A series of testing procedures has been described that helps insure the production of reliable spacecraft.[7] The following is a description of that testing procedure.

Test Objectives:

Development Test: Establish a fundamental behavior pattern upon which a design can be based.

Qualification Test: Verify that the equipment and associated software will meet all specified requirements.

Acceptance Test: Verify workmanship and demonstrate that the equipment functions properly over the range of correctly selected operating conditions.

Prelaunch Verification Test: Performed at the launch site to verify that the spacecraft has sustained no shipping damage and has been properly mated to the launch vehicle.

Interactions with other Subsystems

<u>Mission Planning</u>: The dimensions of the spacecraft in launch configuration limits the mission planner's selection of launch vehicle. Also, the mission planner has selected a flyby mission which greatly simplifies the overall spacecraft configuration.

<u>Science</u>: The scanning and pointing requirements of the scientific instruments requires that the scan platform be positioned in a clear field of view. The scientific instruments must also be provided with shielding from the contaminating space environment. Thermal control must be provided.

<u>Attitude and Articulation Control</u>: The spacecraft inertia and COM, determined by component masses and positions, affects the placement of attitude thrusters and thruster force selection. Thermal control must be provided.

<u>Command, Control, and Communication</u>: The antenna size and placement places restrictions on the placement of the scan platform for clear viewing. The massive computers housed in the bus significantly affect the spacecraft inertia and COM. Also, the computers generate heat that must be rejected from the bus by radiating louvers.

<u>Power and Propulsion</u>: The propellent tank, when fuelled, is the most significant factor in determining the spacecraft inertia and COM. The main propulsive unit must be oriented so that its line of force acts through the spacecraft COM. Thermal control must be provided.

Appendix 1A: Inertia Calculations

The calculation of the individual component inertia's is simplified greatly by idealizing those components into simple geometric shapes. This assumption yields results which are adequate for the purposes of this preliminary design report. Of course this simplified methodology is in no way appropriate for actual inertia calculations of the later stages of design. Another simplifying assumption is that each component is homogeneous in density. Also, for the purpose of this calculation the mass of the bus includes the bus structure, command and control computers, thermal control, and cabling. The following is a list of component idealizations, component inertias, and component COMs. All dimensions are in meters. All inertias are in units of $kg\text{-}m^2$.

Bus (370 kg): Hollow cylinder. L=.35 Ro=.95 Ri=.65 COM=(0,0,0)

$Ix=Iy=M[(Ro^2+Ri^2)/4+(L^2)/12]=126.3$

$Iz=M(Ro^2+Ri^2)/2=245$

Propellent Tank (120 kg empty): Spherical shell. R=1.0 COM=(0,0,-.94)

$Ix=Iy=Iz=2MR^2/3=80$

Antenna (5 kg): Flat disk. R=1.85 COM=(0,0,.7)

$Ix=Iy=MR^2/4=4.3$

$Iz=MR^2/2=8.7$

Antenna Base (45 kg): Solid cylinder. L=.12 R=.95 COM=(0,0,.5)

$Ix=Iy=M[(R^2)/4+(L^2)/12]=10.2$

$Iz=MR^2/2=20.3$

Magnetometer Boom (5 kg): Thin rod. L=13 COM=(7.45,0,0)

$I_x=0$

$I_y=I_z=ML^2/12=70.4$

RTG Boom (5 kg): Thin Rod   L=3   COM=(-.5,-2.45,0)

$I_y=0$

$I_x=I_z=ML^2/12=3.7$

Science Boom (35 kg): Thin rod.   L=1.2   COM=(0, 1.55,0)

$I_y=0$

$I_x=I_z=ML^2/12=4.2$

Scan Platform (111 kg): Prism.   L=.5   W=.3   H=.3   COM=(0,2.2,0)

$I_x=M(W^2+H^2)/12=1.7$

$I_y=I_z=M(W^2+L^2)/12=3.1$

RTG (60 kg): Cylinder.   R=.1   L=1.52   COM=(-.53,-4.71,0)

$I_y=MR^2/2=.3$

$I_x=I_z=M[(R^2)/4+(L^2)/12]=11.7$

Appendix 1B: References

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MISSION MANAGEMENT, PLANNING, AND COSTING

Introduction

Mission Management, Planning and Costing (MMPC) has several responsibilities regarding the unmanned mission to Pluto. A mission timeline, outlining such features as the launch date, impulse points, encounters with planets, arrival at Pluto, and the proposed end of mission date must be furnished. MMPC must also determine a trajectory system so that time and ΔV are optimized. Another responsibility is the selection of the launch vehicle. A vehicle which minimally satisfies the spacecraft's dimensions at launch as well as the mass of the launch configuration is necessary. Furthermore, MMPC must also select the type of mission to be performed at Pluto. The mission should stress simplicity, reliablity and low cost. Lastly, a total costing analysis for the project must be furnished.

The remainder of the MMPC section contains a detailed analysis of the requirements previously mentioned, including trade studies and mission planning effects on other subsystems. The requirements are treated as separate categories where applicable, and each will be discussed individually.

Mission Timeline

On May 4, 2009 (day 0) NASA will launch the spacecraft Copernicus into a low earth orbit (LEO) of 270 km and an eccentricity of 0.00. The spacecraft will then leave the Earth's orbit via an upper stage and begin it's voyage to Pluto. On March 1, 2010 (day 300.4) Copernicus will fire an impulse to prepare for its gravity assist at Jupiter. This gravity assist at Jupiter will occur on February 18, 2012 (day 1019.8). The spacecraft will then be on a trajectory for the planet Saturn, arriving on July 29, 2015 (day 2276.6). Once again, a gravity assist will be made. Copernicus will then travel uninterrupted for about six years until it reaches its target destination, Pluto. The spacecraft will fly by Pluto on December 14, 2021 (day 4607.0). It will then continue on, leaving our solar system, not to return. The end of the mission will occur after the encounter with Pluto on December 14, 2021 (day 4607.0).

During it's flight, Copernicus will be performing correction maneuvers (see Attitude and Articulation Control) when necessary. As they cannot be predicted, no mention of it is included in this time schedule. Figure 2.A shows a timeline view of the mission, from the launch date to the end of mission date.

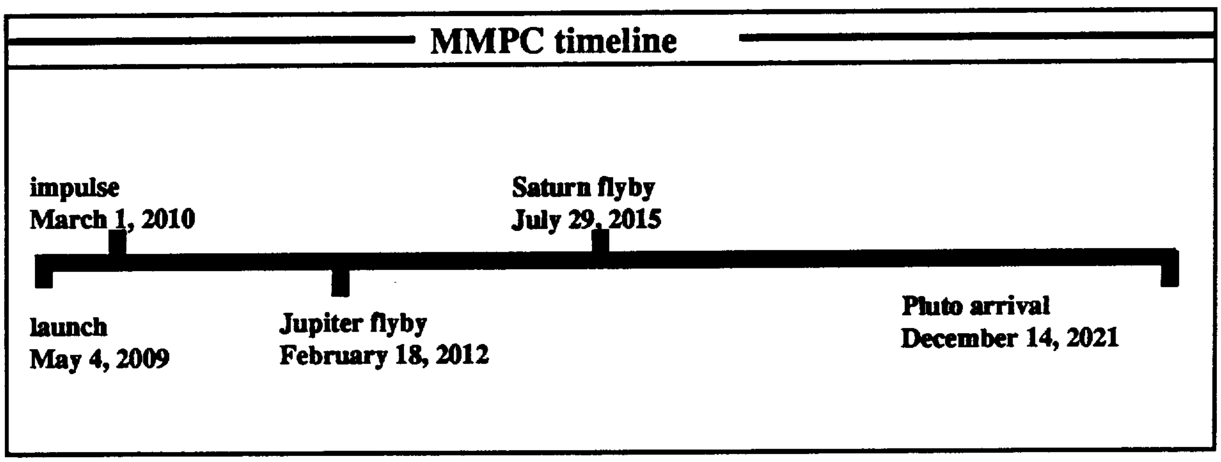

Figure 2A. Mission Timeline

The overall duration of the mission is 12.613 years (4607.0 days). During this time a management program will be in effect. The structure of this program will include a management, control, administration and support staff as well as division representatives[1]. Also, the duration time pertains only to flight of the spacecraft and does not include the planning, research and development and the assembly and testing.

Trajectory Systems

The selection of a trajectory system is perhaps the biggest task for the MMPC subsystem. The spacecraft ideally should arrive at Pluto in a minimum amount of time, while using a minimum amount of fuel. This immediately produces a conflict. A compromise which effectively minimizes both is desired.

The analysis of a trajectory system was performed with computer software. The spacecraft had the requirement that it must be launched sometime in the first decade of the twenty-first century. The spacecraft would have to travel about 33 AU's. A direct flight to Pluto was on the order of 28 years[2]. This was double the desired flight time so efforts to use gravity assists were employed. The first system consisted of using Jupiter as a gravity assist. Much work cut the flight time down considerably to about 15-16 years[2]. However, more planet gravity assists to shorten the flight time were still necessary. The next project involved using Jupiter, Saturn and Neptune for gravity assists. The project was named EJSNP (Earth-Jupiter-Saturn-Neptune-Pluto). The project was aborted in one week. Neptune could not line up properly in conjunction with the other planets, and was requiring too large a ΔV to correct it. So project Pluto began, consisting of an "Earth-Jupiter-Saturn-Pluto" configuration. This cut the flight time down on the order of 13-13.5 years. However, one problem was that it is desirable to leave before or after the first decade of the twenty-first century for Saturn and Jupiter to align properly, preferably early. Another problem is that Jupiter is not to be approached closer than 10 body radii due to large radiation output and Saturn should not be approached closer than 2.4 body radii due to it's rings. The Jupiter restriction was not a problem but Saturn continually required an approach of less than two body radii. The project was switched to "Longshot", using the same bodies as project Pluto but using a launch time at the end of the decade. This allowed Saturn's restriction to be satisfied and produced a flight time of about 12.8-13.2 years. The following graph (Figure 2B) depicts a trade off between ΔV required and time for Operation

Longshot. The final trajectory selection was then determined (Figure 2C.).

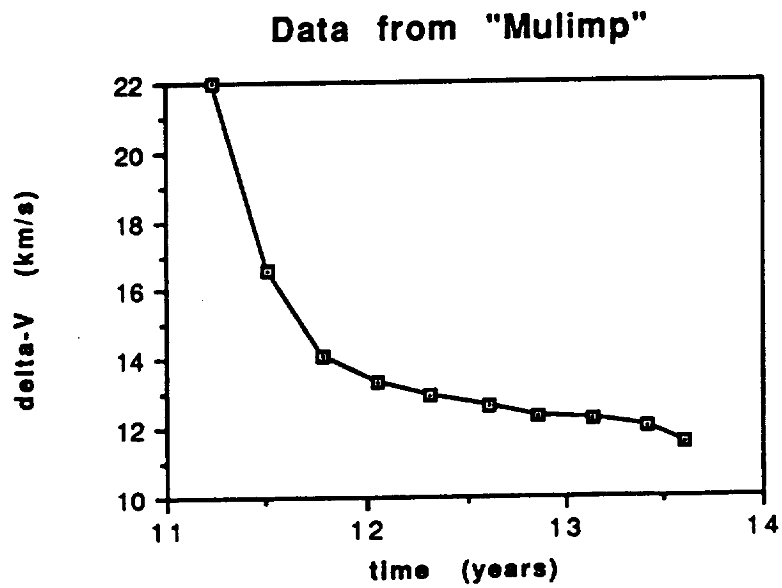

Figure 2B. Mulimp Data

The final selection was optimized to produce an impulse to assist the gravity assist at Jupiter. The mission flight time was finally reduced to 12.613 years. The complete analysis of this mission can be found in the appendix after this subsystem, including but not limited to launch time, ΔV required, and the coordinates of the specific events. The final trajectory is mapped in Figure 2C. Note that planet sizes are not to scale but are shown for illustration purposes.

Another problem is the solar system's asteroid belt. To avoid any possible collision that might result in a mission failure, the impulse fired after departure will provide a ΔV of 0.267 km/sec in the negative z-direction (see Appendix). Another advantage with this trajectory is that it uses all of its fuel (not including the safety factor

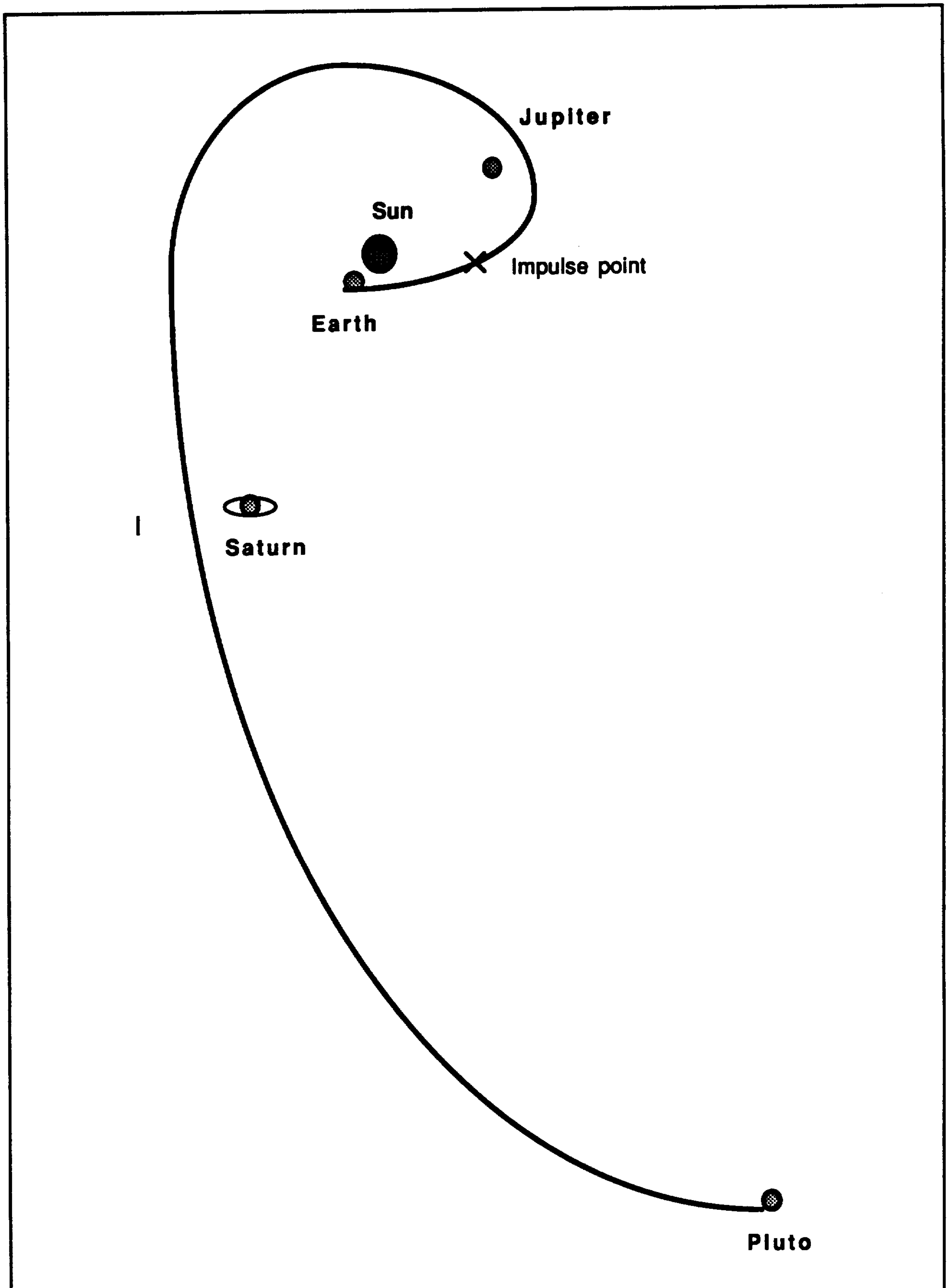

Figure 2C. Copernicus Trajectory

of fuel) early. After 2.798 years, Copernicus will have made its last burn and will travel the remainder of its ten years with the weight of the spacecraft only (excluding attitude control fuel). This is also responsible for its short flight time. The total ΔV required for the mission is 12.371 km/sec. This includes the departure from the Earth's orbit. The ΔV required from the spacecraft's propulsion module is 6.123 km/sec. This can also be found in the Appendix following this subsystem.

Launch vehicle selection

The spacecraft Copernicus requires a launch vehicle to insert it into earth's orbit. The selection of vehicles was limited to United States launch vehicles. The launch vehicle would have to be able not only to reach orbit, but it was also desired to use a configuration that would let the spacecraft escape the earth's gravity and to begin it's mission.

The launch vehicle must satisfy the spacecraft's weight including fuel and launch packing. A factor of safety of at least 10 percent was also desired. The companies that were considered were Martin Marietta, General Dynamics (GD)/Space Systems, McDonnell Douglas, and Boeing. Initially, GD/Space System's Atlas G was selected. However, as more fuel was added, the spacecraft's weight increased and the minimum performance payload necessary became 2730 kg and the Atlas G could no longer meet the requirement[4]. The spacecraft's pre-launch configuration had the dimensions of a cylinder of radius 3.7 m and a height of 5.0 m (see "Structures"). These dimensions could be employed on most launch vehicles and was not a primary concern.

The launch vehicle finally selected consisted of the vehicle and an upper stage. The launch vehicle selected is the Titan T-34D, by Martin Marietta. This, used in combination with the Centaur D1-T, could handle a payload weighing up to 5910 kg[4]. This exceeds the minimum requirement easily. However, the Centaur upper stage is built by GD/Space Systems and the Centaur D1-T is a modified version of the Centaur, designed specifically for the Titan T-34D.

The Titan T-34D uses a solid propellant while the Centaur D1-T uses LOX/LH_2. Both vehicles are environmentally safe and pose no conflicts regarding the safety of the launch.

Mission Type

The type of mission selected is the result of a lengthy trade analysis. The types of missions were divided into three categories: flyby, lander and orbiter. A flyby class mission was identified as any mission which did not perform any thrusting at Pluto. A lander mission was defined as the landing of any item on Pluto's surface. Lastly, an orbiter mission involved using a burn to obtain an orbit about the planet for a given length of time. Of the three classes, only the flyby and the orbiter missions were highly analyzed.

A lander mission involved sending a spacecraft to a planet of which there is little knowledge of. Historically, a lander mission follows an initial study of the planet. For a lander mission to be effective, an accurate idea of what is to be accomplished should be known. It would be senseless to send a lander to Pluto without first knowing what areas of the planet interest us. Also, the difficulties of uncertain areas including the gravity, composition, surface conditions and temperatures possess too high a risk factor for such a mission. Furthermore, the cost of carrying out a lander mission to Pluto might as much as double that of a flyby.

Initially, ideas for an orbiter mission were assembled. An orbiter could perform many experiments, and would also allow a longer encounter time at Pluto. Also, the mission was to incorporate a needle probe to penetrate the surface of Pluto and to examine samples. However, the ΔV required was high (a burn of 9.0-11.5 km/sec was required to insert the spacecraft into orbit[2]). Also, further research posed yet a bigger problem: Pluto's moon, Charon.

This was of no major concern at first. However, since Charon's mass exceeds 4 percent of Pluto's mass[6], the two bodies behave as a binary system. This system would make an attempt to orbit Pluto very difficult. Essentially, a three body problem must be solved. Another idea would be to orbit in a "figure eight" configuration. Also, while Charon's sphere of influence is estimated at 7000 km , Pluto can retain a satellite up to an estimated 5000000 km[6]. The mission tended to lean toward flyby class at this point. To orbit Pluto, burns would likely be needed for stable equilibrium. This suggests that the orbit duration would be short (finite), and a finite orbit duration did not warrant the increased cost of fuel required for orbit insertion. The mission to send an orbiter to Pluto was finally aborted. A mission to flyby Pluto was decided.

A flyby mission is the least expensive to build, test and fly. The components needed for the mission are considerably less than that of an orbiter class mission, making it a simpler design and more reliable. Furthermore, a flyby mission has an attractive ΔV (see "trajectory system"). While a flyby mission has less of an opportunity to gather information, it still provided adequate instrumentation, including imaging equipment to make an initial survey of the planet. Lastly, the spacecraft would ideally leave the solar system permanently. The spacecraft will have drawings on it's buss including a picture of man, as well as the location in our solar system in the Milky Way galaxy in the event of an encounter with any intelligent life. Only a flyby mission would allow this to occur.

Costing

The costing of the spacecraft includes the cost of not only the design and research leading to the construction of the vehicle, but the ground support operations of the lifetime of the mission. A detailed analysis of the costing can be found in the appendix following the end of this subsystem. The costing estimation used in this report is the "model estimation" method. This primarily involves assigning a number of labor hours to each section of the spacecraft. The labor hours are in turn converted into labor cost and the labor cost is finally related to the total cost. The total cost of the spacecraft is $999,443,600 dollars in terms of the 1977 fiscal year.

Another estimation technique is the concept of inheritance. The model estimation technique uses the masses of the individual systems but gives no consideration to the design and research development of the systems. Inheritance involves assigning each system to one of five classes:

Class One:	Off-the-Shelf/Block Buy
Class Two:	Exact Repeat of Subsystem
Class Three:	Minor Modifications of Subsystem
Class Four:	Major Modifications of Subsystem
Class Five:	New Subsystem

Any components from class one will benefit from the previous design while class five receives no benefits whatsoever. By incorporating inheritance into the model estimation technique, the final cost will be effectively estimated. Assume that four spacecraft will be built for costing purposes.

Appendix 2A.

Costing-

Section1:

This contains the determination of direct labor hours (DLH) and recurring labor hours (RLH). The standard format is either $x*(N*M)^y$ or exp(x+y*N*M), where N is the number of spacecraft and M is the mass in kilograms. Note DLH and RLH are given in thousands of hours.

NRLH = DLH - RLH

	Inheritance Class
<u>Structure and Devices</u>	
DLH = 1.626*(2*285)^0.9046 = 947.1	
RLH = 1.399*(2*285)^0.7445 = 264.0	
NRLH = 683.1	3
<u>Thermal Control, Cabling & Pyrotechnics</u>	
DLH = exp*(4.2702 + 0.00608*4*30) = 148.4	
RLH = 3.731*(4*30)^0.6082 = 68.6	
NRLH = 79.8	3
<u>Propulsion</u>	
DLH = 56.1878*(4*120)^0.4166 = 735.6	
RLH = 1.0*(4*120)^0.9011 =260.7	
NRLH = 474.9	3
<u>Attitude & Articulation Control</u>	
DLH = 21.328*(4*49)^0.7230 = 968.8	
RLH = 1.932*(4*49) = 378.7	
NRLH = 590.1	2

Telecommunications

DLH = 4.471*(4*20)^1.1306 = 633.9

RLH = 1.626*(4*20)^1.1885 = 297.1

NRLH = 336.8 2

Antennas

DLH = 6.093*(4*5.1)^1.1348 = 186.6

RLH = 3.339*(4*5.1) = 68.1

NRLH = 118.5 2

Command & Data Handling

DLH = exp (4.2605 + 0.02414*4*49.7) = 8600.1

RLH = exp (2.8679 + 0.02726*4*49.7) = 3972.6

NRLH = 4627.5 3

RTG Power

DLH = 65.300*(4*60)^0.3554 = 458.0

RLH = 7.88*(4*60)^0.7150 = 396.6

NRLH = 61.4 3

Line-Scan Imaging

DLH = 10.069*(4*36.5)^1.2570 = 5291.5

RLH = 1.989*(4*36.5)^1.4089 = 2228.4

NRLH = 3063.1 2

Particle & Field Instruments

DLH = 25.948*(4*39.0)^0.7215 = 991.8

RLH = 0.790*(4*39.0)^1.3976 = 917.8

NRLH = 74.0 2

Remote Sensing Instruments

DLH = 25.948*(4*44.5)^0.5990 = 578.2

RLH = 0.790*(4*44.5)^0.8393 = 61.2

NRLH = 517.0 2

Section 2:

This section analyzes the Development Project - Support Functions and the Flight Project. PPL is in units of pixels/line. MD is the mission duration in months and ED is the encounter duration in months.

PPL = 1024
MD = 151.2
ED = 4.0
Σ DLH(hardware) = 19540
NRLH = DLH

<u>System Support & Ground Equipment</u>

DLH = 0.36172(ΣDLH)^0.9815 = 5887.3

<u>Launch + 30 Days Operations & Ground Software</u>

DLH = .09808(ΣDLH) = 1916.5

<u>Imaging Data Development</u>

DLH = 0.00124(PPL)^1.629 = 99.4

<u>Science Data Development</u>

DLH = 27.836(non-imaging science mass)^0.3389 = 124.7

<u>Program Management/MA&E</u>

DLH = 0.10097 (ΣDLH all categories)^0.9670 = 602.5

<u>Flight Operations</u>

DLH = (ΣDLH/3100)^0.6*(10.7*MD + 27.0*ED) = 5208.8

<u>Data Anlalysis</u>

DLH = 0.425*(DLH Flight Operations) = 2213.7

Section 3:

Total Costing:

This section incorporates inheritance into the costing. Costing for class 2 =1.00(RLH) + 0.2(NRLH). Costing for class 3 = 1.00(RLH) +0.75(NRLH). Since both equations represent labor hours, they must be converted to dollars.

LH = labor hours = (1.0-Z)*NRLH + RLH
Z = percent cost reduction
LC = labor cost
TC = total cost

Cost Category	LH	LC to TC
Structure & Devices	776.3	26975.0
Thermal Control, Cabling & Pyrotechnics	128.4	4369.8
Propulsion	616.9	23511.7
Attitude & Articulation Control	496.7	17671.9
Telecommunications	364.5	12205.8
Antennas	91.8	3169.1
Command & Data Handling	7443.2	227894.7
RTG Power	442.7	13375.4
Line-Scan Imaging	2841.0	108225.8
Particle and Field Instruments	932.6	33624.8
Remote Sensing Instruments	164.6	5760.3
System Support & Ground Eq	5887.3	191053.5
Launch+30 days Ops & Ground S/W	1916.5	65969.6
Image Data Development	99.4	3565.5
Science Data Development	124.7	6344.0
Flight Operations	5208.8	176571.4
Data Analysis	2213.7	79155.3
Totals	29749.1	999443.6

Total Cost of the Copernicus mission: $ 999,443,600

Equations Pertaining to MMPC-

TC = total cost = (100%-Z) NRC + RC
see costing section of appendix for individual component equations.

Final Trajectory Orbital Elements-

On the following page is an excerpt containing the orbit elements for the final design trajectory. This contains various data, including but not limited to flight time, ΔV required, and the Cartesian coordinates of significant encounters.

Final Trajectory Orbital Elements

```
*************************************************************************
CASE TITLE: OPTIMIZATION REPEATS WITH ADDED IMPULSE(S)

FLIGHT TYM:         4607.0 DAYS          12.613 YEARS
      MODE: TOTAL DV OPTIMIZATION
            TERMINAL FLYBY(UNCONSTRAINED)

            CENTRAL BODY IS SUN          EARTH ECLIPTIC & EQUINOX [illegible]
   LAUNCH:     C3=   75.887   C3PRIME=   75.887
              VHL=    8.709       DLA=  -15.724      LONG= [illegible]
 IMPULSES:  NO.     TYM(JD)      BODY     DVX(KPS)  DVY(KPS)  DVZ(KPS)  DV(KPS)

             1  2454956.192     EARTH          ORBIT DEPARTURE            [illegible]
             2  2455256.552        NO     0.000    -0.004   [illegible] [illegible]
             3  2455976.040     JUPIT    -2.264    -5.209   [illegible] [illegible]
             4  2457232.910     SATUR     0.000     0.000   [illegible] [illegible]
             5  2459563.192     PLUTO     0.000     0.000   [illegible] [illegible]
                                                  TOTAL DV(KPS)       [illegible]

IMP POINTS: NO.       DATE        DAYS      X(AU)     Y(AU)     Z(AU)   R(AU)

             1  2009 MAY  4        0.0   -0.7312   -0.6914  [illegible] [illegible]
             2  2010 MAR  1      300.4    2.1709   -0.2101  [illegible] [illegible]
             3  2012 FEB 19     1019.8    2.5548    2.4912  [illegible] [illegible]
             4  2015 JUL 29     2276.4    1.5391   -8.2952  [illegible] [illegible]
             5  2021 DEC 16     4607.0   11.3309   22.0007  [illegible] [illegible]
 SUN DATA:  NO.       VHP(KPS)   DEC(DEG)   RA(DEG)   PHAZ(DEG)  [illegible] [illegible]

             1*         8.709    -15.724   210.369    37.501  [illegible] [illegible]
             2          0.000      0.000     0.000     0.000  [illegible] [illegible]
             3*         6.024     -1.008   165.306   102.220  [illegible] [illegible]
             4*        15.766    -21.420   330.072     9.774  [illegible] [illegible]
             5*        13.796     64.004    81.230     4.971  [illegible] [illegible]
               * = BODY EQUATORIAL VHP COORDINATES
 S/A DATA:  NO.      VHC(KPS)    RP(BR)   DAM(DEG)  FCI(DEG)  [illegible] [illegible]

            3( 0)      17.214    10.000    77.961    77.717  [illegible] [illegible]
            4( 0)      15.765     2.454    53.507    53.571  [illegible] [illegible]
 RUN DATA:  NO.     TYM GRAD  POSXGRAD  POSYGRAD  POSZGRAD

             1    0.797E-02 0.000E+00 0.000E+00 0.000E+00
             2    0.927E-03-0.112E+01-0.217E+00-0.265E 02
             3    0.127E-01 0.000E+00 0.000E+00 0.000E+00
             4   -0.209E-01 0.000E+00 0.000E+00 0.000E+00
             5    0.000E+00 0.000E+00 0.000E+00 0.000E+00
            NS=       1     1     7     0     0    20
           NRI=       0     0     0     0
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COMMAND, CONTROL, AND COMMUNICATIONS

Computer Control

Copernicus, like most spacecraft, must perform a variety of functions at precise times with unerring accuracy. In order to do this, an on-board computer system is necessary. The computer system must control three main areas, the attitude and articulation subsystem (AACS), the flight data subsystem (FDS), and the computer control subsystem (CCS). A schematic layout of the computer system is shown in Appendix 3A. The computer will be made of three separate, freestanding but interacting computers, controlling the three areas mentioned. This system is modeled after the system on board the Voyager spacecraft.

The FDS computer is responsible for all of the flight data received during the lifetime of the spacecraft. All of the data from the science platform as well as all the periodic status reports of the spacecraft are fed into this computer, where it is assimilated, reduced and passed on. The FDS computer will be a 16 bit x 8192 word computer, as on the Voyager, and will interact with the rest of the computer system as well as the science platform and most other instruments for status reports.

The AACS computer is responsible for keeping Copernicus going in the right direction, with the correct orientation in space. All tracking data is fed into the AACS computer and it decides if a readjustment burn is necessary to correct its trajectory. Every reorientation of the spacecraft, to allow burns or communications, is timed and the AACS computer knows when to command the burns and precisely how long to burn. The AACS computer will be an 18 bit x 4096 word computer. This provides ample room for all of its programming needs.

The CCS computer is also an 18 bit x 4096 word computer. Most of the permanently stored programs are kept in this computer. If necessary it can completely reprogram both the AACS and the FDS. This provides a vital redundancy factor for the spacecraft computer system. Should the CCS need reprogramming, that would need to be done from Earth. All information to be sent to Earth and all incoming information from Earth goes through the CCS computer before moving on to the other computer subsystems, the antenna, or other areas of the spacecraft.

The three components of the overall computer system interact fully and all feed into a central storage unit, as shown in Appendix 3A. The data storage unit has a 400 kilobits per second(kbps) record rate, which will be able to handle all of the incoming data from the various computer subsystems. It also has five different playback rates, 100, 50, 25, 12.5, and 6.25 kbps. This wide range will handle all of the needs of the computers, the science platform and telecommunications.

Communications

Communications back and forth between Copernicus and Earth is essential for proper mission accomplishment. Copernicus needs to relay information such as status reports, scientific data and imagery back to Earth, while the command center on Earth needs to be able to send commands to the spacecraft to have it perform certain functions such as execute a burn, change course or take a picture. While most of the necessary commands for Copernicus will be stored in the computer system it is still necessary for communications to be able to reach the spacecraft.

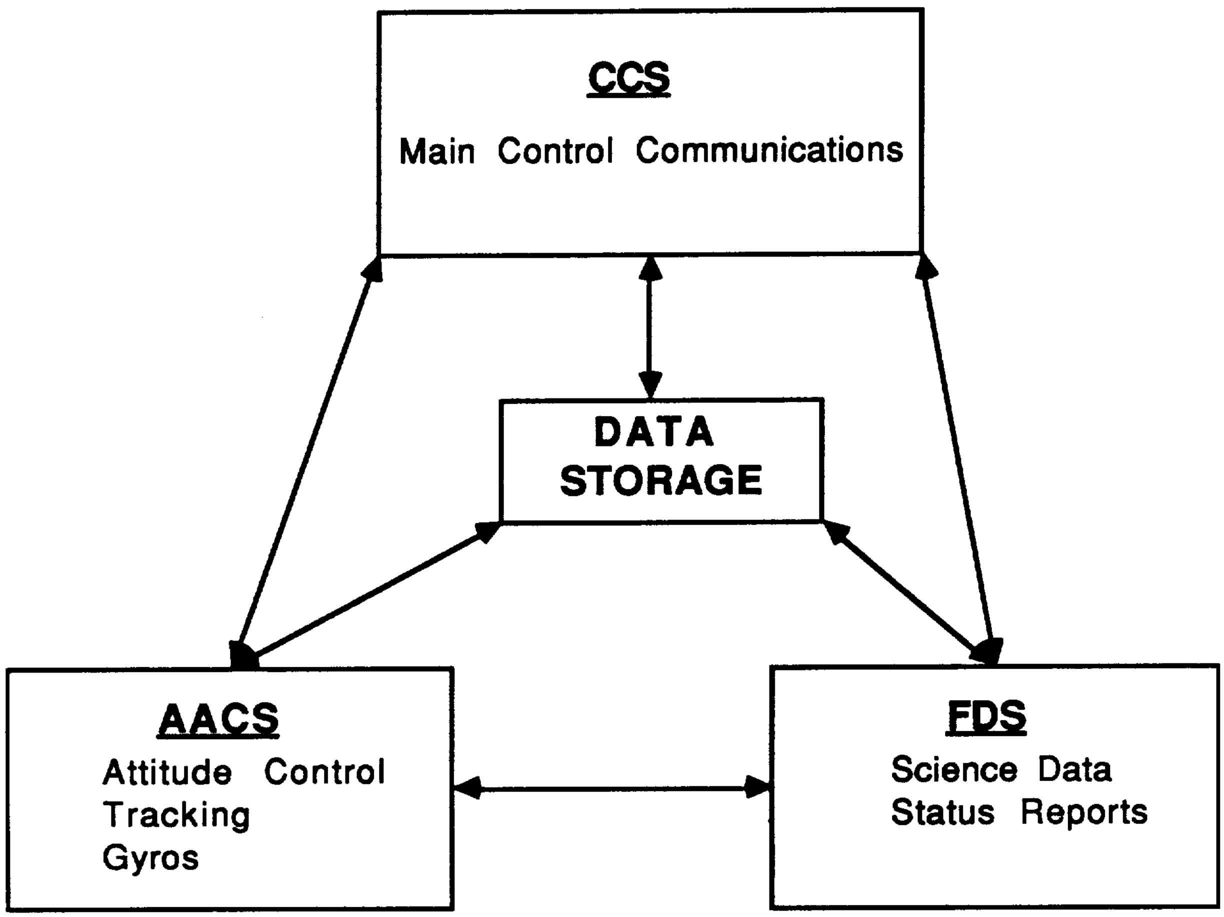
An antenna is the instrument used to perform the necessary transmission and collection of data. Copernicus' antenna is a standard parabolic dish that focuses the radio waves it intercepts to a central receiving unit, or broadcasts the radio waves onto the dish which sends them back to Earth.

There are two general radio wave frequencies used in deep space telecommunications, S-Band and X-Band. The X-Band is generally preferred due to its higher frequencies, which have less interference problems, and it will be used for Copernicus. The X-Band uplink (Earth to space) frequency is 7.161 GHz while the downlink (space to Earth) is 8.414 GHz. There are many factors that affect the transmission energy before it reaches its destination. These factors are summed up by the equation in Appendix 3B. Most of these factors are losses that reduce the energy from transmission to reception.

The size of the antenna is the driving factor in the calculation of necessary power. Large antenna sizes have larger gains, so less power is needed to achieve a required receiving power. Our antenna has a significant mass and keeping the mass to a minimum is important, so we can not allow our antenna to become too large. Another factor involved in the sizing of the antenna is the fact that it must fit within our launch vehicle. This means that the antenna must either be kept small or be collapsible, and much more complicated. In order to keep the configuration of Copernicus simple and less costly a solid antenna was chosen. It will be 3.7 meters in diameter. This provides Copernicus with a small, lightweight antenna that fits within the launch vehicle but is still capable of making necessary transmissions with little energy (app. 25 W).

The positioning of the antenna is vital in mission accomplishment. The antenna must point towards Earth if communications between Copernicus and Earth are to occur. Generally, though, the propulsion for the spacecraft points out of the back of the spacecraft, towards Earth. The antenna and the propulsion package will be on opposite ends of the spacecraft. For most of the beginning of the voyage, the antenna will be useless because of the fact that Copernicus will be in its burn stage. After the primary burn stage is complete, Copernicus will rotate 180°, allowing full communications. During the flight, if a burn using the main propulsion is needed, Copernicus must again be rotated 180°.

Appendix 3A. Control Flowchart



Appendix 3B. Communications Equation

$$P_R = P_T L_T G_T L_{TP} L_S L_A L_P L_{RP} G_R L_R$$

P_R=Power Received
P_T=Power Transmitted
L_T=System Losses in Transmitter
G_T=Transmitting Antenna Gain
L_{TP}=Pointing Loss of Transmitter
L_S=Free Space Losses
L_A=Atmospheric Attenuation
L_P=Polarization Loss Between Antennas
L_{RP}=Pointing Loss of Receiver
G_R=Receiving Antenna Gain
L_R=System Losses in Receiver

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POWER
AND
PROPULSION

Overview

The power system aboard the vehicle utilizes inherently reliable components. Only materials and techniques available before 1999 are to be used in the final fabrication of the system. The system design lifetime is sufficient to carry out the mission, allowing for a reasonable safety margin. Under normal mission conditions, the power system is fully autonomous. If necessary, new commands can be transmitted from the ground station on Earth. Performance, simplicity, and low weight and cost are stressed in design tradeoffs.

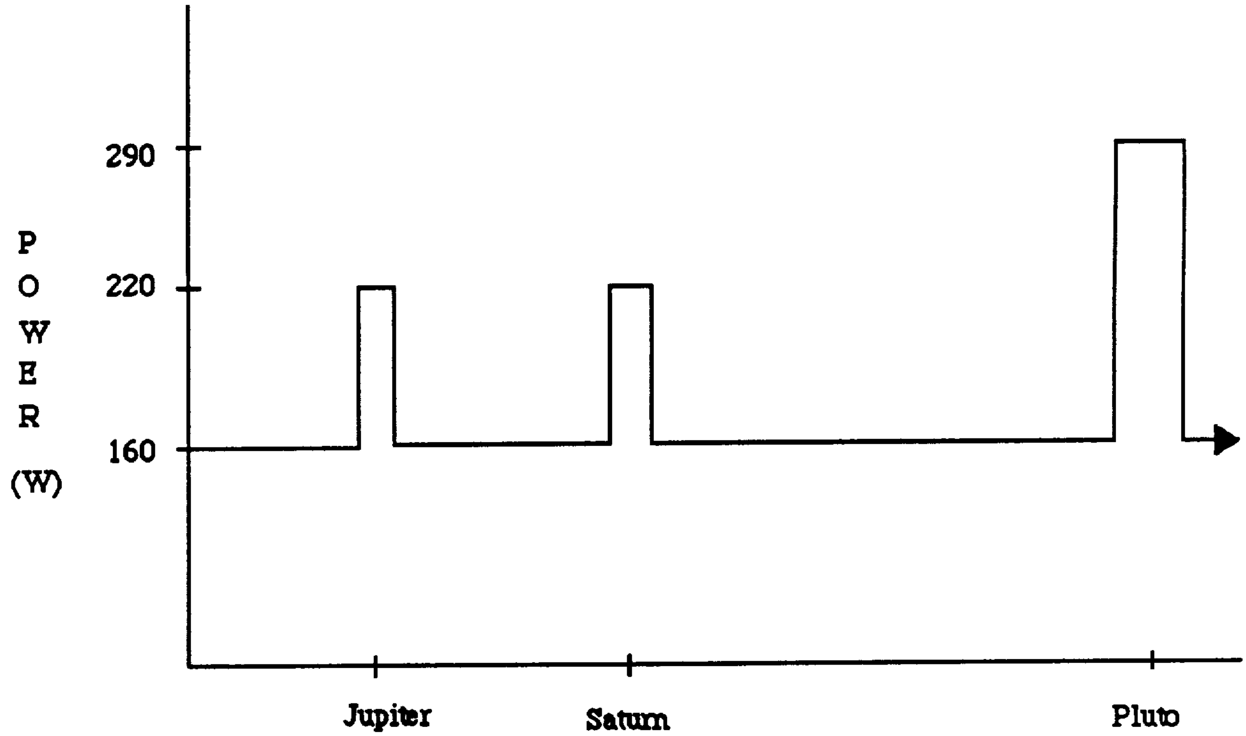
The main power source is a Modular Isotopic Thermoelectric Generator (MITG). With the flyby of several planets, the power requirements will change with respect to the mission timeline. The modularity of this component makes it ideal for use in this mission. Releasing power in small scaled amounts, this unit efficiently meets the power needs of the spacecraft at all times during the mission.

There exist socio-ecological problems in the use of the MITG, problems shared with all isotopic thermoelectric generators. Containing plutonium oxide, debris from these units would be extremely dangerous in the event of launch mishap. These are legitimate concerns and have been taken into consideration of the overall design. For a mission of this duration, however, it is infeasible to incorporate any other type of system.

Table 4-A. Power Requirements

system/component	power requirement
AACS	40 W
Science	130 W
Structure	
thermal control	19.6 W
pyrotechnics	2.4 W
CCC	
computer	24.7 W
data storage	23.2 W
antenna	25 W
Power	25.2 W
Total:	289.9 W

The maximum power required by the system is approximately 290 W. The total power supplied by the MITG is approximately 310 W, sufficient for the load requirements. The maximum power levels will only be reached during planetary flyby. Here the bulk of the scientific instrumentation will consume approximately 60 W of power. The imaging equipment will only be utilized at the encounter with Pluto, requiring an additional 70 W. The modification of power supplied will be autonomously controled by the computer.



PLANETARY FLYBY ENCOUNTERS
Figure 4-A.

The earlier planetary encounters require power increases for only a few days centered about the flyby date. In the case of Pluto, the imaging process requires weeks of the increased power level. An insignificant power of 2.4 W is needed for pyrotechnics at separation of the vehicle from the upper stage.

Component Selection

The MITG design was conceived by Fairchild Space and Electronics Company. They have developed several unit sizes ranging from output levels of 260 W to approximately 300 W. Satisfying the power requirement for the spacecraft , the 13 slice generator has been selected.

A redundant circuit design for both the dual busbars and network has been selected to decrease the chances of failure due to micrometeorite impact. Parallel fuses are incorporated on each load to provide redundancy. The electric circuit is located outside the generator housing, minimizing the probability of shorts-to-ground problems. Incorporating field-cancelling circuit modules, scientific instrumentation on the spacecraft will not be affected by induced magnetic fields from the MITG.

The generator consists of 13 independent slices each supplying approximately 24 W at 28 V. Each thermoelectric slice contains four plutonium oxide pellets supplying a total of 250 W of thermal power. A series of eight thermoelectric modules per slice convert the thermal power, given off by the fuel pellets, into electric power for the spacecraft. The plutonium oxide is contained in an iridium clad surrounded by an impact shell. Thermal insulation, consisting of carbon bonded carbon fibers, protects the fuel pellets from under or over-heating. The whole assembly is protected by an aeroshell, designed to maintain its structural integrity at extremely high temperatures. This design uses four radiator fins situated at the corners of the unit, optimizing heat dissipation as well as weight.

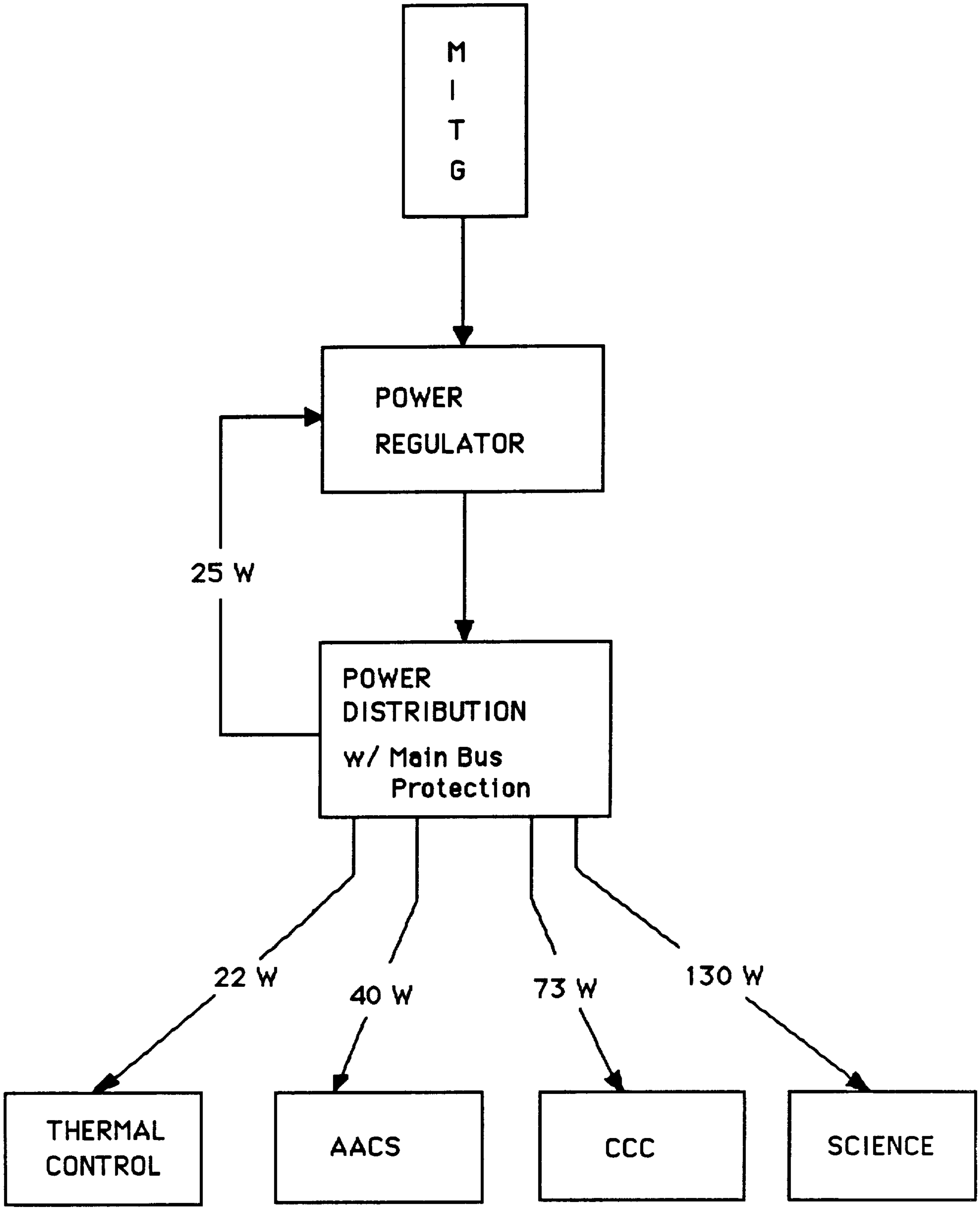


Figure 4-B. Power Breakdown

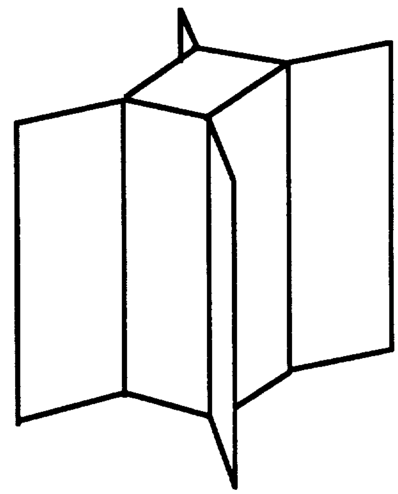


Fig. 4-C. Modular Isotope Thermoelectric Generator

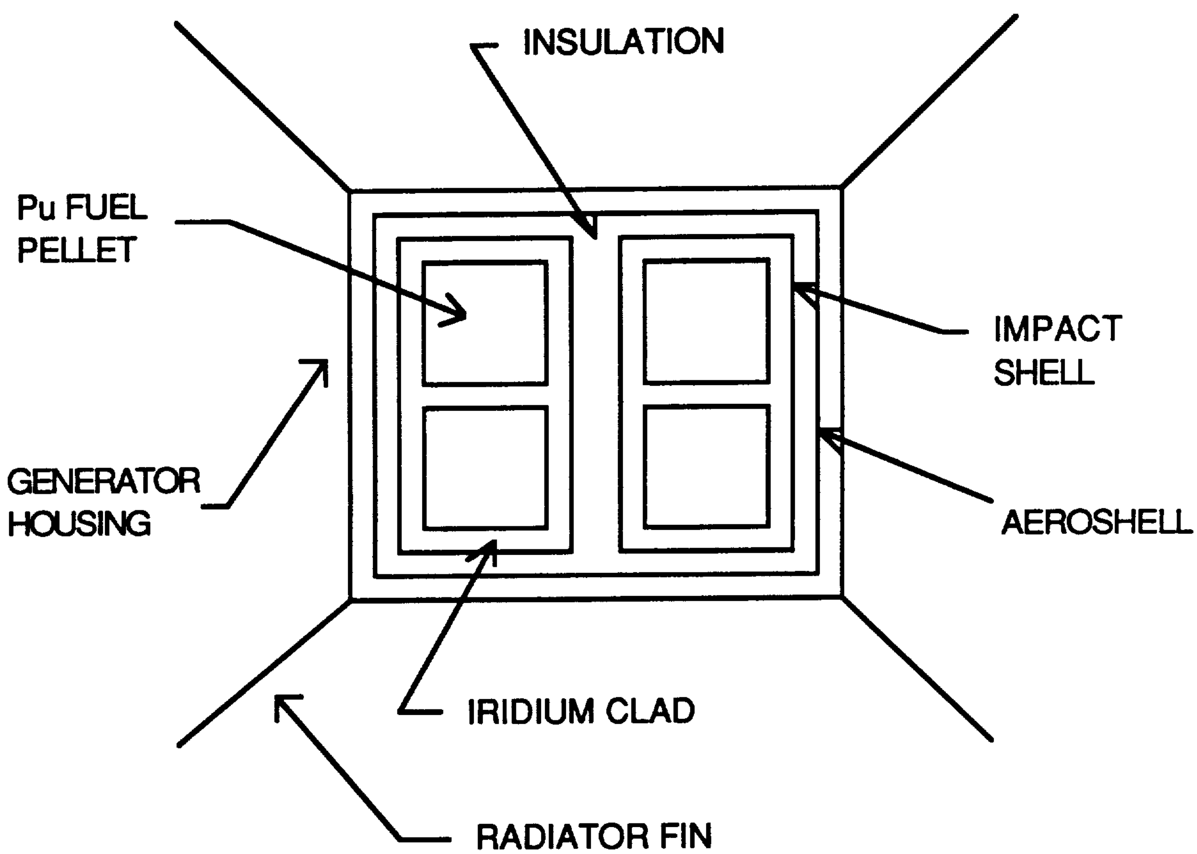


Fig. 4-D. Thermoelectric Slice

Overview

The propulsion system for the vehicle is characterized by simplicity and reliability. Components incorporated in the system have been flight tested extensively, meeting with proposal requirements on availability before the year 1999. The propulsion unit as well as the fuel storage have design lifetimes sufficient to carry out the mission, allowing for use of the thrusters for unexpected mid-course maneuvers. The system relies on autonomous control by the onboard computer. Performance, weight, and cost have been optimized in design tradeoffs.

The fuel used in this system is augmented hydrazine. Similar to conventional hydrazine, it is space storable for long periods of time. Considering the longevity of this mission, storability is essential. Because it is a monopropellant fuel, oxidation systems are not needed, lowering cost and weight. Generally systems of this type are capable of specific impulses of 200 to 250 seconds. With the use of augmented hydrazine, values of 300 seconds specific impulse can be obtained. Advantages of augmented hydrazine include low plume contamination and no surface contamination, problems which could interfere with the normal operation of the spacecraft and scientific instrumentation on board.

The main thrusters will burn twice during the mission. These two burns will provide the spacecraft with a total ΔV of 6.1 km / s. The first burn required is a small mid-course impulse, taking place approximately ten months after launch. The next burn is at Jupiter flyby, approximately two years later. This schedule provides for a smaller probability of error in the propulsion system since all the major burns occur in the first three years. The remaining amount of fuel, used by the attitude and articulation thrusters, will be approximately 5 % that of the initial supply.

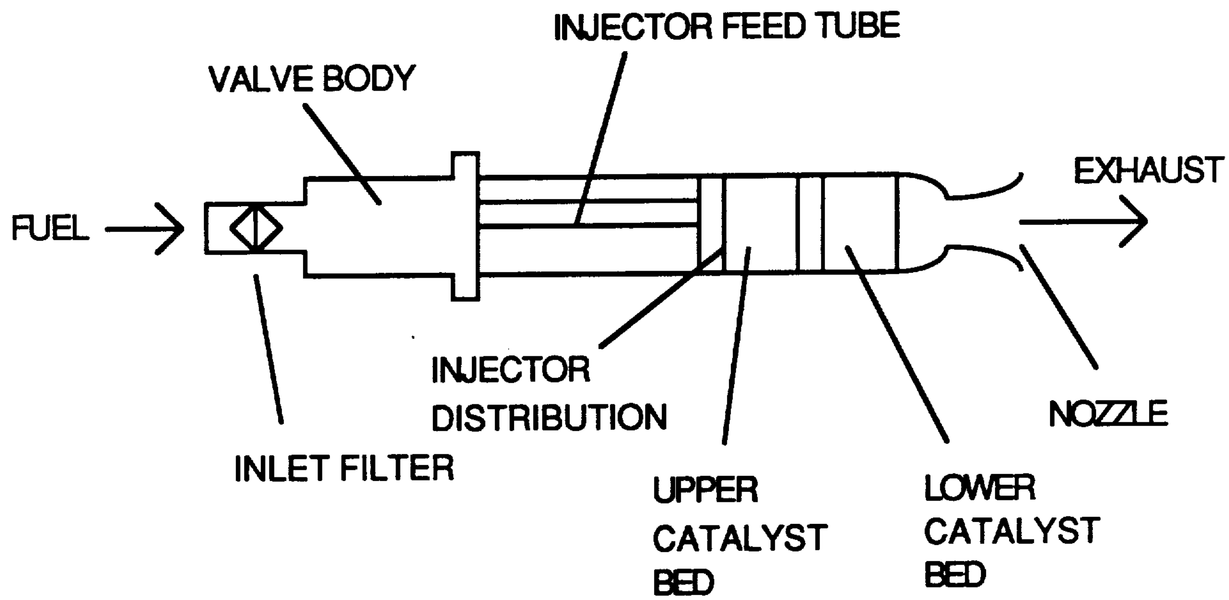


Figure 4-E. Hydrazine Thruster

The fuel storage tank is characteristic of the bladder design, eliminating the need for a pressurizing system. As the fuel is consumed the bladder folds in on itself providing the thrusters with a steady supply of fuel during burns. Since the attitude and articulation thrusters also use the hydrazine fuel, the storage tank can be shared between the two systems. Fuel from the tank travels through an inlet filter, which removes all foreign particles from the fuel stream. From there, it is driven through an injector feed tube and into the injector distribution element. The fuel then passes through the catalyst bed where it is ignited chemically. Heaters are situated around the catalyst bed for the chemical reaction to be carried out properly. Exhaust gasses then escape out of the nozzle providing the spacecraft with the necessary thrust.

Appendix 4A. Equation for Propulsion Subsystem

$$\Delta V = g_o \; I_{sp} \; \ln \left(m_i / m_f \right)$$

ΔV = change in velocity

g_o = constant for gravity

I_{sp} = specific impulse

m_i = initial mass

m_f = final mass

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AAE 241 Class Notes ~ Power Subsystem

AAE 241 Class Notes ~ Propulsion Subsystem

SCIENCE INSTRUMENTATION

Introduction

This section describes the scientific subsection of project Copernicus. This includes a science time line, the planned experimentation, and the equipment needed to complete the mission. The selection of experiments was based on present day scientific objectives for information gathering of the outer planets. Individual instrument systems were compared and selections were made based on experimental need. In addition, the requirements and constraints of NASA's Request For Proposal (RFP) were obeyed.

Voyage to Pluto and Charon

The long voyage to Pluto and Charon will allow an excellent opportunity for Copernicus to gather information on the galaxy. This time will not be wasted. During every phase of the journey, experimentation will take place.

Earth-Jupiter Cruise Phase

After initial Earth orbit and spacecraft deployment have been established, the science mission will begin in earnest. Once out of Earth orbit the scientific equipment will be tested and calibrated through relay with mission scientists on Earth. Later in the journey such fine tuning will not be possible. Copernicus will spend the majority of its time in interplanetary space, at these times science will act in cruise mode. During cruise phases, fields and particles experiments will be employed. Distant stars will be targeted for observation and data recording. Information will be gathered and relayed to Earth approximately every 0.5 AU.

Jupiter Encounter Phase

As Jupiter nears the instrumentation and experimentation will convert to encounter mode. The scan platform will be turned to focus directly on Jupiter. Approximately 80 days and 80,000,000 km before the closest approach to Jupiter, Copernicus's imaging equipment will come to life. Over the next seven weeks, the narrow angle camera will take visual information of the whole planet. A series of color filters on the camera will also be employed. At this time, the infrared and ultraviolet spectrometers along with the photopolarimeter will be taking whole planet data.

As Copernicus approaches 30,000,000 km from closest approach, the transmitter will begin sending information at encounter data rate. At this time, the wide angle camera and its color filters will be engaged. The fields and particles experiments will also be placed in encounter mode. Specifically, they will investigate the transition from the region of space dominated by the solar wind to that of Jupiter's magnetosphere.

As closest approach nears, the equipment on the scan platform will take advantage of the change in phase angle, from low phase angles to high, to observe any differences in information due to the phase angle change. During Jupiter pass by, the Earth will be eclipsed from Copernicus which will allow an excellent opportunity for mission scientists observe the effects of the Jovian atmosphere on the communications signal. This radio science information could be used to draw conclusions about the composition and height of the Jovian atmosphere.

As Copernicus leaves it will pass through Jupiters shadow which will allow ultraviolet inspection of the atmospheric upper layer composition. Also, long exposure imaging of Jupiters night side will take place. As the probe continues out the fields and particles experiments will investigate the extended tail of the magnetosphere. Transmission will return to cruise data rate 40 days after closest approach.

Jupiter-Saturn Cruise Phase

Upon entering the Jupiter-Saturn cruise phase, science investigations will return to primarily fields and particles. Special attention will be paid to the gradual changes in the character and temperature of the solar wind. Particles experiments will emphasize the cosmic ray environment. During this phase, the annual solar conjunctions allow radio science the opportunity to investigate the solar corona. As communication signals transverse the solar corona mission scientists can measure the coronal electron density.

Saturn Encounter Phase

The Saturn encounter will progress as did the Jupiter encounter. The only difference being the emphasis on Saturns rings. Imaging will begin 80 days out, fields and particles experiments and transmission rates begin encounter mode 30 days out, and Copernicus returns to cruise mode 40 days after closest approach. The information gathered from the Jupiter and Saturn encounters can be compared to data obtained from the Voyager missions. Any differences found could be very useful in understanding our changing planets and galaxy[4].

Saturn-Pluto Cruise Phase

In the final interplanetary cruise phase Copernicus will investigate the proton component in the distant solar wind plasma. It will also measure the intensity, composition, and differential energy spectrum of galactic cosmic rays. These experiments are very important, as no other spacecraft has taken this final route.

The power and data rate requirements of the science subsection are shown in time line format in Figure 5A. This clearly portrays the distinct peaks of power use and transmission requirements during the planetary encounters. The power capabilities and communication needs are adequately met by the Copernicus spacecraft. Figure 5B indicates the individual instruments used in each phase of the mission. The instruments were selected for each phase to maximize the data gathering and to minimize the power drawn and the data transmitted.

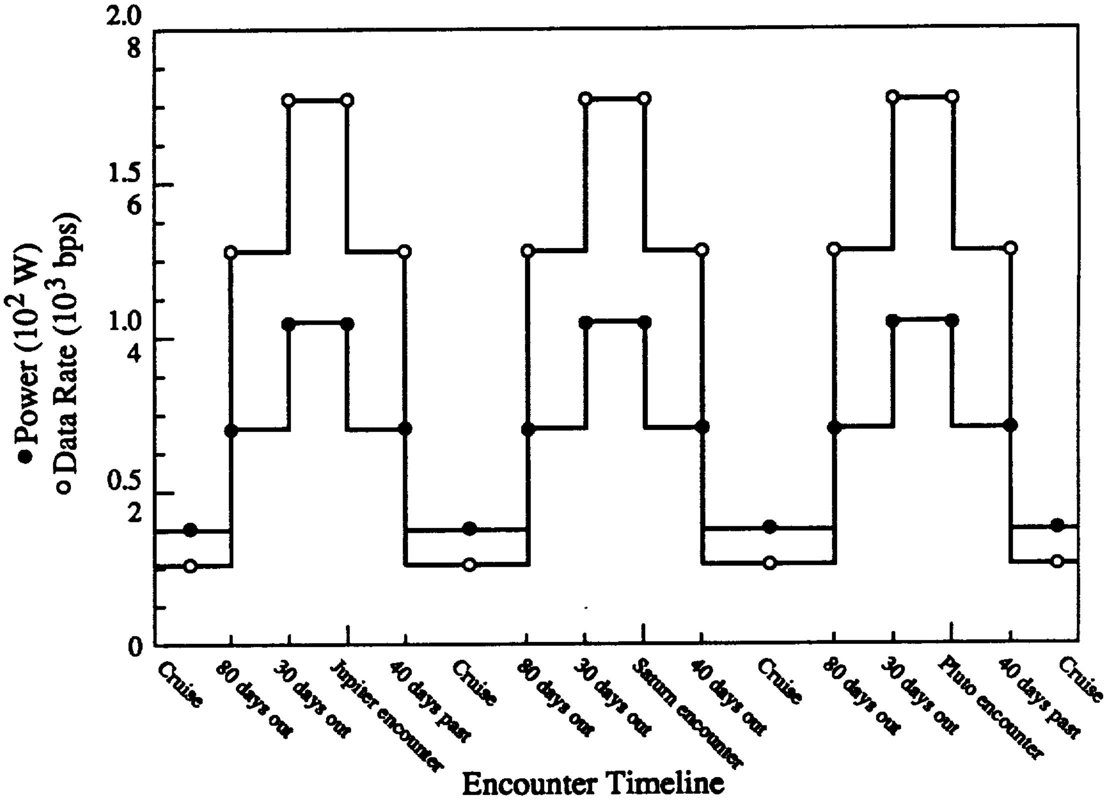

Figure 5A Power and Data Transmission Timeline

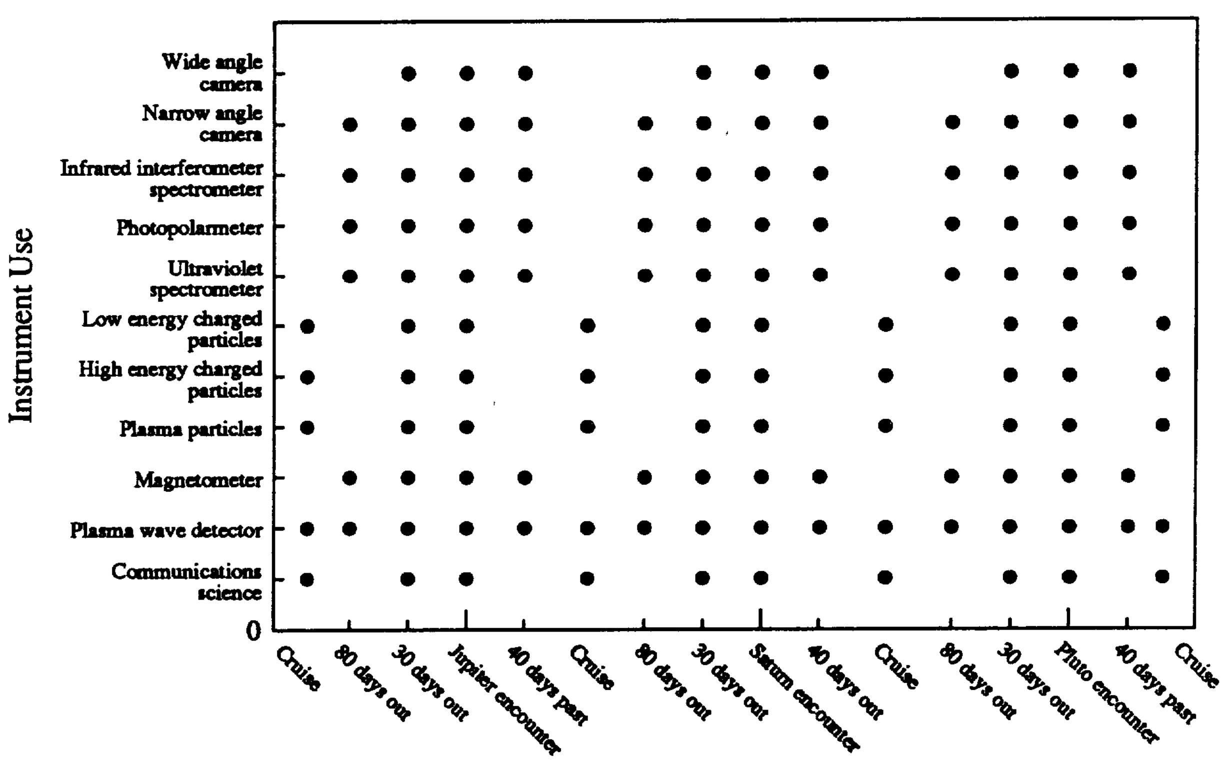

Figure 5B Instrument Use Timeline

Pluto and Charon Encounter

The mission culminates with the investigation of Pluto and its satellite Charon. Scientific objectives for the two bodies were based on those from the National Academy of Sciences objectives for the outer planets[5]. Experiments and investigations specific to Pluto and Charon were developed that would fulfill the needs of the scientific community. These experiments in approximate order of importance can be seen in Table 5A[2,6,7]. Many of the investigations have specific subexperiments.

Experiments in Approximate Order of Importance
1. Total Mass and Density • Map the surface albedo distribution • Investigate ice to rock ratio • Investigate composition and hydration state
2. Radius and Oblateness • Find global maps of Pluto and Charon • Investigate hydrostatic shape changes • Map solid body shapes
3. Atmospheric Composition • Investigate atmospheric induced limb darkening effects
4. Gravitational Harmonic Coefficients
5. Shape and Strength of magnetic Field at Several Radii
6. Pattern and Magnitude of Heat Flux, Surface Temperature, and Heat Balance at Various Latitudes and Phase Angles
7. Shape and Intensity of the Tail of the Magnetosphere or of the Cavity in the Solar Wind
8. Local Anomalies • Investigate possible dark spots and rings

Table 5A Experimental Listing

The experiments will proceed in a similar manner to the encounters of Jupiter and Saturn. However, because the system being investigated is a two body system, care will need to be taken with respect to time management. The scan platform will need to be rotated to allow adequate time to gather data from both Pluto and Charon.

As Copernicus approaches Pluto and Charon, the imaging equipment will begin the investigations of radius and oblateness. It will begin to compile images that will be used to create global and solid body maps, and to investigate any hydrostatic shape changes. These maps will be used to help determine the radius and oblateness of both bodies. As the probe nears, the infrared interferometer spectrometer will be used to investigate thermal emissions, composition of thermal structure, and heat balances. This data will be collected over a variety of phase angles. The information, along with the imaging data will help to map the surface albedo distribution, investigate the ice to rock ratio, find the pattern and magnitude of heat flux, surface temperature, and heat balance at various latitudes and phase angles of both Pluto and Charon. While still on approach, Copernicus will accumulate data with its magnetometer. Information from the magnetometer will aid in determining each bodies gravitational harmonic coefficients and the shape and strength of their magnetic fields at several radii. The photopolarimeter will investigate the physical and chemical properties of Pluto and Charon. This information, along with data from the infrared interferometer spectrometer and the imaging equipment, will help to determine the composition, mass, and density of both bodies.

As the spacecraft passes through its closest approach, the particles experiments will convert to encounter mode. In this mode they can gather a variety of important information. The high energy particle detector will measure electrons and cosmic rays, while the low energy particle detector investigates particles in the planetary magnetosphere. The plasma particle detector will determine plasma flow direction and the plasma wave detector will study the wave and particle interaction in the dynamics of the magnetosphere. All this information will be used to model the shape and intensity of the tail of the magnetosphere or of the cavity in the solar wind.

While the probe is eclipsed from Earth, additional investigations will be made. The imaging equipment will focus on Plutos limb and terminator region. Data acquired can be used to determine the atmospheric induced limb darkening effects. Communications tracking of the probe can aid in finding Plutos gravitational harmonic coefficients and the strength of its magnetic field.

As Copernicus sails into the outer galaxy its investigations will not end. Possibly it could investigate the heliopause. It will continue to send data from our galaxy back to Earth.

Equipment Selection

The design features a wide variety of imaging, spectroscopy, and fields and particles instruments. All equipment was selected from existing hardware used on the Cassini, Galileo, and Voyager missions. This was done to minimize cost while keeping a high level of information accuracy and reliability.

Imaging Science Subsystem (ISS)

The Copernicus probe will encounter a wide variety of targets and range of observing distances. Therefore, two separate cameras will be used in the ISS, a Narrow Angle Camera (NAC) and a Wide Angle Camera (WAC). In this way, Copernicus can provide two different scales of image resolution and coverage.

The two cameras are framing Charge Coupled Device (CCD) imagers. The charge couple device design is a square array of 1024 × 1024 pixels, each pixel is 12 μmeters on a side. They differ primarily in the design of the optics: the NAC has a focal length of 2000 mm and the WAC has a focal length of 250 mm. Both cameras have a focal plane shutter of the Voyager/Galileo type, and a two-wheel filter changing mechanism derived from the Hubble Space Telescope. Both cameras have deployable dust covers. To minimize mass, power, and cost, the two cameras will not be completely independent - they will share a common electronics module. This module services both cameras, and contains the digital part of the video signal chain, power supplies, mechanism drivers, command and control logic, and the digital data compressor[3].

Key parameters of the ISS:

	Narrow Angle	Wide Angle
Camera Type	Framing CCD	Framing CCD
Optics Type	Ritchey-Chretien	Refractor
Focal Length	2000 mm	250 mm
Focal Ratio	f/10.5	f/4.0
Resolution per pixel	6 μrad	48 μrad
Field of View	0.35° square	2.8° square
Spectral Range	200-1100 nm	350-1100 nm
Spectral Filters	22	14
Heater Unit	Strip heaters	Strip heaters

Infrared Interferometer Spectrometer (IIS)

This instrument consists of an infrared radiation telescope, two Michelson interferometers for evaluating spectral data, and a radiometer for measuring total body reflection. The IIS will be used to measure planetary thermal emissions, surface composition, and thermal structure. It will accomplish this by measuring reflected solar radiation and heat balances[1,4].

Photopolarimeter

The photopolarimeter gathers information on surfaces or particles by observing how they scatter light. To accomplish this the photopolarimeter must take measurements over a variety of phase angles. This data can be evaluated to find the physical and chemical properties of planetary atmospheres and surfaces. The intensity and polarization of light are measured in 10 narrow bands from 0.41-0.945 microns, including areas where methane and ammonia strongly absorb radiation[1,4].

Ultraviolet Spectrometer (UVS)

The ultraviolet spectrometer operates in two distinct modes: airglow and solar occultation. During Copernicus's cruise phases, the UVS will operate in airglow mode. It will observe the sources of extreme ultraviolet radiation in the galaxy. As the probe enters an encounter phase and passes by a planet, the ultraviolet spectrometer will convert to solar occultation mode. In this mode the instrument will study solar light and the effects a passing planets atmosphere has on it. The UVS covers a 0.115-0.43 micron spectrum and views with a 0.1° slit width. The ultraviolet spectrometer can detect nitrogen, sulfur, and atomic hydrogen and oxygen. Microprocessor control provides flexibility. The UVS can fix at one wavelength and look for intensity changes during a scan, or it can rapidly step through wavelengths for a full spectrum over a broader area - or some combination in between[1,4].

Particles Investigations

The particles studies consist of three distinct instrument investigations. They are a Low Energy Charged Particle (LECP) detector, a High Energy Charged Particle (HECP) detector, and a Plasma Particle (PP) detector. The LECP detector operates with two objectives: measure particles in planetary magnetosphere and to detect low energy charged particles in interstellar space. It accomplishes its objectives by measuring particle source, composition, energy spectra, flux intensity, and favored particle direction. The HECP detector is similar to the low energy charged particle detector, however it measures particles by charge, mass, energy, and arrival direction. The LECP and HECP work with a combined range of 0.020-55 million electron volts for ions and 0.015-11 million electron volts for electrons.

The plasma particle detector consists of two Faraday cup plasma sensors and three mass spectrometers. Its objective is measuring the plasma in the solar wind and in planetary magnetospheres. It is also responsible for finding the plasma flow direction. The PP detector studies plasma by detecting its velocity, density, and pressure. This device measures the energy range of electrons and positive ions from 1.2-50,400 electron volts. The

Faraday cup plasma sensors collect the plasma data, while the three mass spectrometers are included to identify the composition of ions[1,4].

Fields Investigations

The instruments that fall under the fields category are the magnetometers and the plasma wave detector. The magnetic fields investigations employs four magnetometers. This investigation uses two sets of two triaxial fluxgate magnetometers. One set is of low field, the other high field. These magnetometers measure planetary magnetic fields. They measure with a range of 0.00032-0.16384 gauss.

The plasma wave detector will be used to study wave and particle interaction in the dynamics of a planets magnetosphere. The detector measures changes in electric and magnetic fields. The electric and magnetic fields can be measured separately over ranges of 5 Hz. to 5.6 MHz. and 5 Hz. to 160 KHz., respectively[1,4].

Table 5B shows the scientific mission at Pluto/Charon of each instrument Copernicus will be carrying. All equipment will be heated with a combination of strip heaters and passive athermalization with invar and aluminum structures.

Instrument Layout

Instruments will reside in one of three locations aboard the spacecraft. The magnetometer boom, the scan platform, or the scan platform boom. The scan platform and its boom, along with the magnetometer boom were located so as to maximize their distance from each other and from the Radio Isotope Thermal Electric Generator (RTG).

Scan Platform

The scan platform will house the instruments that specifically need to be pointing at the target they are investigating. It will be extended out from the Copernicus by a folding boom. The platform itself will have two axis of freedom about which to rotate. This will

EQUIPMENT	INVESTIGATION CONCERNS	
Imaging	Radius Oblateness Global maps Solid body maps Limb darkening	Map surface albedo Ice to rock ratio Mass Density Terminator region
Infrared Interferometer Spectrometer	Thermal emissions Composition of thermal structure Heat balances Heat flux Mass	Map surface albedo Ice to rock ratio Surface temperature Composition Density
Magnetometer	Harmonic coefficients	Magnetic fields
Photopolarimeter	Physical, chemical properties Mass	Composition Density
HECP Detector	Measure electrons Tail of magnetosphere	Measure cosmic rays Cavity in the solar wind
LECP Detector	Particles in magnetosphere Cavity in the solar wind	Tail of magnetosphere
Plasma Particle Detector	Plasma flow direction Cavity in the solar wind	Tail of magnetosphere
Plasma Wave Detector	Particle interaction Cavity in the solar wind	Tail of magnetosphere
Ultraviolet Spectrometer	Atmospheric composition	

Table 5B Instrument Investigations

minimize the maneuvering required from the spacecraft. The instruments on the scan platform include the narrow angle and wide angle cameras and their electronics, the infrared interferometer spectrometer, the photopolarimeter, the ultraviolet spectrometer, and the plasma wave detector. The equipment will be placed together and bore sighted with the narrow angle camera. By placing the instruments in a cluster, the strip heaters can serve more than one instrument, thereby minimizing power use and cost. Because the equipment will be bore sighted on the narrow angle camera, mission scientists will have an image corresponding to data collected from the other scan equipment.

Requirements are placed on the movement of the scan platform by the science instrumentation, specifically the imaging equipment. The platform can rotate with a maximum slew rate of 0.33° per second. At this rate the instruments with the exception of the imaging equipment can be accurately used after a settling time of 45 seconds. However, if the cameras are to be employed a settling time of 288 seconds is required. The equipment on the scan platform also places a limit to the maximum maneuver rate of the spacecraft. The maximum allowable maneuver rate of Copernicus while performing experiments, except imaging is 0.033° per second. The maneuver rate while imaging drops to 0.00972° per second. Another requirement for the scan platform is its pointing accuracy. The platform must be high precision with pointing accuracy of at least 2 mrad with 1 mrad knowledge and stability of 10 mrad in 0.5 seconds and 100 mrad in 100 seconds. Figure 5C represents a view of the scan platform and its equipment[3,8].

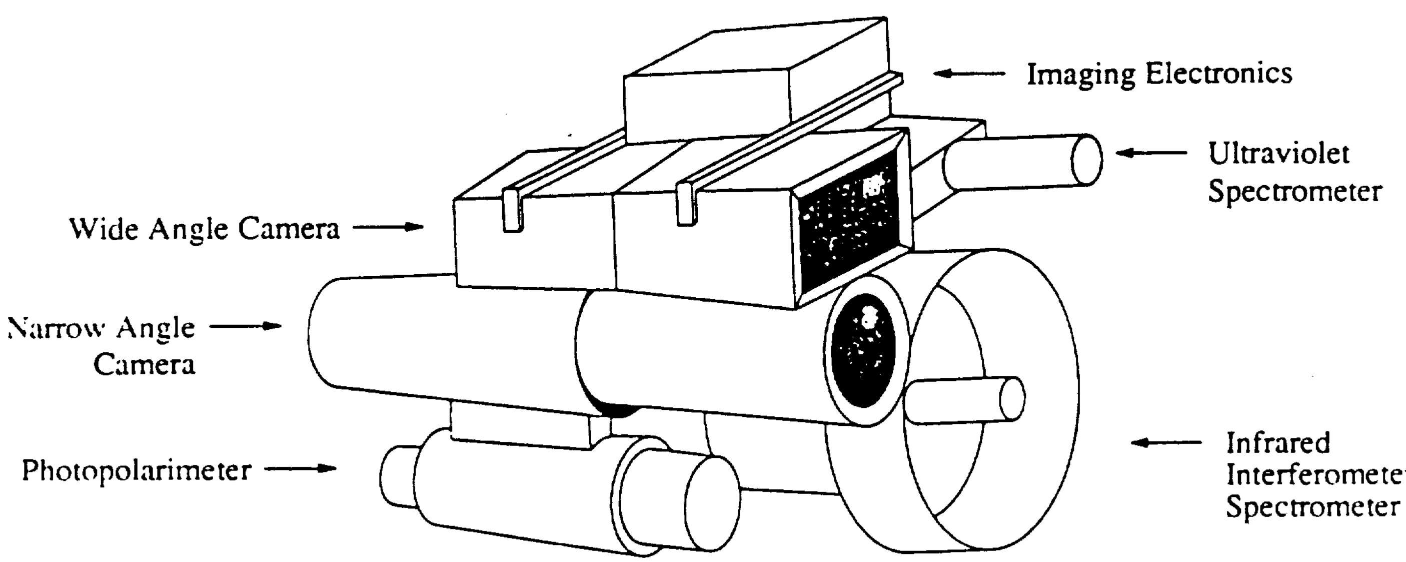

Figure 5C

Scan Platform Boom

The scan platform boom is a convenient location to place the particles instruments. It is away from the spacecraft and allows undisturbed flow through of the interstellar environment. The boom will house the low and high energy charged particle detectors and the plasma particle detector.

Magnetometer Boom

This 13 meter long boom will remove its low field magnetometers from interference with the other science equipment. The magnetometers will be the only instruments placed on this boom. The high field magnetometers will be place on the boom near its attachment to the spacecraft. One low field magnetometer will be located half way down the boom, the other placed at the farthest end.

Table 5C is a listing each instrument and its mass, power requirement, data transmission rate, and location on the probe[4,8].

INSTRUMENT	MASS (kg)	POWER (W)	DATA RATE (bps)	LOCATION
Imaging	36.5	29.0	3850	Scan Platform
Infrared Interferometer Spectrometer	18.5	12.0	500	Scan Platform
Photopolarmeter	13.0	13.0	450	Scan Platform
Ultraviolet Spectrometer	13.0	13.0	450	Scan Platform
LECP Detector	9.0	16.0	450	Scan Boom
HECP Detector	13.8	16.5	450	Scan Boom
Plasma Particle Detector	9.9	8.1	450	Scan Boom
Magnetometer	4.9	5.8	400	Magnetometer Boom
Plasma Wave Detector	1.4	1.6	200	Scan Platform

Table 5C Instrument Data

Conclusion

The mission to Pluto and Charon can only be completed cost effectively by a spacecraft whose science section maximizes accurate data gathering and the number of target investigations, while minimizing mass, power consumption, and complexity. The Copernicus probe meets these requirements.

APPENDIX 5A
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ATTITUDE
AND
ARTICULATION
CONTROL

Introduction

The task of the Attitude and Articulation Control System (AACS) is to control the attitude of the spacecraft. This requires pointing the high gain antenna toward the Earth and/or Sun, pointing the trajectory correction thrusters in any direction, providing control authority during the rocket engine burns, performing science maneuvers, and pointing the scan platform.

These control requirements are very challenging because of the complex and time changing parameters the Copernicus will encounter. Initially, there is the change in mass at separation from the launch vehicle, and then the changes in mass during mid-course correction and orbit burns. Propellant slosh is and wobble amplifications are also factors.

These requirements and the time-varying parameters dictate a complex set of AACS sensors and actuators controlled by a high performance computer, and that a great deal of on-board autonomy be present in the AACS. Also there are weight and power constraints that put stringent requirements on the electronic components. A mission objective is to prevent single-point failures from jeopardizing the mission. This forces redundancy of the critical components and requires internal fault protection logic to control that redundancy.

Without doubt, accurate attitude control of the Copernicus is imperative to mission success. This section describes the attitude control of the Copernicus spacecraft during the entire mission, giving detailed descriptions of the components and methods used in designing the AACS.

Attitude Control Modes

The attitude of the Copernicus is achieved through the use of a set of celestial sensors, a set of inertial sensors, an onboard digital computer, and a set of hydrazine thrusters. The Copernicus will be three-axis stabilized due to the science requirement for a scan platform and the lower cost compared to a dual spin design. Three-axis stabilization also permits extended viewing of selected targets, thus permitting a larger number of individual measurements or a longer integration time for increased sensitivity per measurement than can be achieved with a spin stabilized spacecraft unless it has a de-spun platform.

On account of the length of the mission, the Copernicus must be able to function autonomously for a large amount of its travel time. A basic guideline is that the spacecraft (S/C) be able to operate for at least one week without ground intervention without loss of more than one science instrument or loss of more than one-half the engineering telemetry and the S/C must be left in a commandable state. Therefore it is imperative that the control computer have various fault detection and correction actions when the S/C subsystems experience certain failures, and be able to maintain correct attitude control during these times[6].

A software estimation process has been derived to determine the best spacecraft position, rate, and acceleration estimates in the presence of noise and disturbance processes. Based on these estimates the attitude of the spacecraft is corrected by activating the appropriate hydrazine thrusters. The algorithm for determining the best spacecraft position and rate is described in Appendix 6A [1].

During cruise, the normal response to a fault is to "safe" the S/C in a specifically oriented attitude. However, during critical mission phases, the on-board systems must reconfigure the Copernicus in such a way as to maximize the probability of completing critical sequences (such as burn and science maneuvers). To accomplish various maneuvers necessary in reorienting the Copernicus, a commanded turn capability is implemented. A turn in any of the three axes is accomplished by the insertion of a bias in the control loop during inertial cruise.

Scanning Platform and Pointing Control

The mounting of a science scan platform at the end of a science boom permits the physical tie-down of its mass during launch, provides for mass balancing of the RTG's for spacecraft center of mass control, and maximizes the unobstructed solid angle through which the remote sensing instruments can be pointed. This platform holds all of the science instrumentation and sensor and control components, which have accurate pointing requirements, thereby eliminating many sources of error that have existed on prior spacecraft. Clearly, the pointing performance of this platform is critical to the success of the mission.

Typical pointing requirements for a high precision scan platform (HPSP) are shown in Table 6-A. These requirements are primarily driven from the requirements of the cameras, and apply to each of

Table 6-A. Pointing Requirements

High Precision Scan Platform Requirements	
Inertial Pointing Control	2.0 mrad (0.11°)
Inertial Pointing Knowledge	1.0 mrad (0.06°)
Inertial Pointing Stability (during 0 to 17.5 mrad/sec slew)	10 μrad/0.5 sec 100 μrad/100sec

the two required axes of articulation. These requirements fall well within the requirements for the entire Copernicus mission. The dynamics of the platform boom can be excited by both basebody motion and platform slews. The choice of an appropriate scan actuator which controls this platform, and compensates for disturbances, will be described next.

Scan Actuator

A key element in the mission is the high precision scan platform. On this platform a number of instruments are mounted, including several cameras and the star tracker and gyro used for S/C attitude control. Clearly, the pointing performance of this platform is critical to the success of the mission. The central consideration of a scan actuator can have an impact on the design of the entire spacecraft.

A direct drive actuator with a platform mounted momentum compensation wheel is selected for the Copernicus. This actuator is selected on the basis of net effect on spacecraft mass, required power, cost, expected pointing performance, necessary control complexity, suitability to mission, operational considerations, and ability to accomodate changes in the mission or spacecraft. It is assumed that all actuators considered met the spacecraft reliability and lifetime requirements.

Table 6-B compares four models of possible actuators, including a momentum compensation harmonic drive (MCHD), direct drive, harmonic drive (HDA), and two-motor actuators. It can be seen that

Table 6-B. Scan Actuator Comparison

Criteria	Direct Drive	Two-Motor	MCHD	HDA
Reliability	Least Risk	Acceptable	Acceptable	Unacceptable
Mass	27 KG	51 KG	50 KG	31 KG
Total Power Peak/ Steady State	8W/6W	17W/12W	11W/8W	10W/6W
Performance	1 μrad	16 μrad	7 μrad	N/A
Heritage	Galileo	Pathfinder	Breadboard	Halley Intercept

overall the direct drive actuator is the best choice, with the bonus that it's been space tested on the Galileo.

The reason for the momentum compensation wheel is that a savings in attitude control propellant can result in an overall savings of spacecraft mass for missions requiring a large number of platform slews, such as Copernicus. Thus when the scan platform accelerates in azimuth, the motor-mounted wheel with the required inertia ratio will accelerate in the opposite direction. The elevation axis works the same way. So ideally the spacecraft body will not sense the platform articulation disturbance torques.

The direct drive actuator is the simplest of the configurations considered. It consists of a brushless DC motor mounted at the gimbal joint. Torque is applied directly by the motor to the platform and a reaction torque is applied directly to the basebody[5].

Star Tracker

The development of charge-coupled device (CCD) optical sensors has made it possible to construct high-performance star and target trackers for spacecraft. They offer high resolution, dimensional stability, and both geometric and photometric linearity. The ASTROS-II (Advanced Star/Target Reference Optical Sensor) tracker currently being developed at the Jet Propulsion Laboratory is scheduled to be launched on the Comet Rendezvous Asteroid Flyby mission. This tracker uses the RCA 501 DX CCD, has integral microprocessors to control the data acquisition, make image position calculations, and provide an effective interface to the pointing control computer.

Table 6-C compares available star sensors. The ASTROS-II is based on the ASTROS built for flight on a series of shuttle-based ultraviolet astronomy missions. The revised design will be tailored to requirements of the Copernicus mission. The ASTROS-II has the following capabilities:

a) Tracks several stars simultaneously for attitude reference (up to 5 stars per field).

Table 6-C. **Star Tracker Comparison**

Characteristic	CS-203	Canopus	ASTROS	ASTROS-II
Mission	VRM	Voyager	Shuttle	Copernicus
Field of View	4.6° wide	9° x 36°	2.2° x 3.5°	11.5°x11.5°
Drift Rate (°/sec)	0.2-1.0	N/A	<.1	<.5
Internal Redundancy	Yes	No	No	Yes
Dimensions (cm)	17x24x18	29x13x11	50x25x20	25x16x16
Mass (kg)	5.5	4.3	2.8	8
Power (w)	7	4.5	38	11

* VRM - Venus Radar Mapper

b) Follows rapidly moving , time-varying , extended targets during a close flyby or rendezvous.

c) Determines the limb position and orientation of a nearby target.

d) Develops image data for ground-based target searches during target approach.

e) Tracks both stars and extended targets and provides optical navigation data for the mission.

f) Mass, power, volume, and environmental compatibility with the Copernicus mission.

These qualities make the ASTROS-II an optimal choice for the Copernicus mission. The unit will be internally redundant and therefore the specifications listed in Table 6-C make it a substantially better choice than all others. The tracker will be located on the HPSP along with the scientific instrumentation[2].

Laser Gyro

The attitude of the Copernicus in three-space is measured by a new technology gyro based on fiber optics, Fiber Optic Rotation Sensor (FORS). Nearly 100 years ago, it was discovered that light, along with conventional gyroscopes, could provide gyroscopic information. The time it takes light to traverse a circular pathway depends on whether the pathway is stationary or rotating. The time difference can measure the amount of rotation[7].

The FORS design uses a single 5 mW GaAlAs laser to input light, divided and injected, into both ends of a 3 to 20 km long fiber waveguide wrapped around an 18 cm coil. After the light has passed through the fiber waveguide, it is recombined and detected. This concept is based on the Sagnac interferometer principle. The phase angle between the two light beams is dependent upon coil rotation rate, direction, number of turns of the fiber, and area enclosed[3].

There will be two sets of three of these gyros for redundancy. The use of this type of gyro results in a planetary gyro with ten times improved drift rate over today's conventional gyros. With the absence of moving parts, no gas discharge tube, and no short term wearout mechanisms, the operating lifetime is well within the mission requirements for Copernicus.

The fabrication processes are relatively inexpensive. The absence of moving parts and close similarity to electronic microcircuit fabrication allow this. The recurring cost of these new planetary gyros is less than one-third of today's conventional gyro cost. The mass, power, and volume will also be

Table 6-D. Gyro Comparison

Unit	Drift Rate(°/sec)	Angular Resolution	Power (w)	Mass (kg)	Volume (cm^3)
FORS	2×10^{-4}	0.005 arcsec	10	10	16400
DRIRU-II	3×10^{-3}	0.05 arcsec	22	11	16236
CG-1300 Laser	7×10^{-3}	1.4 arcsec	18	18	5740

less than present gyros. Table 6-D compares the FORS and two other currently available gyros. The entire gyro component will be placed on the science scan platform for optimal accuracy[3].

Reaction Control System (Thrusters)

The Reaction Control System (RCS) of the Copernicus consists of twelve 1N thrusters located in four clusters about the center of mass of the spacecraft, illustrated in Figure 1-C (Structures Section). The RCS is a monopropellant hydrazine system which has fuel supply lines running from the main propellant bladder. The thrusters are similar to the Voyager design and act as couples. They provide attitude control torques and thrust for small engine maneuvers and trajectory correction maneuvers (TCM), but the main propulsion engine provides most of the control thrust during impulse burns and large maneuvers. The use of four clusters with three thrusters each provides redundancy, designating main and backup sets of thrusters which can be used for control about specific axes. An example of the designated control setup is shown in Figure 6-A and Table 6-E.

The thruster is designed to provide 0.95N thrust and 300s specific impulse at propellant inlet pressure of 24.6 kgf/cm^{2}a to meet the requirements for Copernicus. The thruster has a 60:1 expansion ratio conical nozzle. Thrust level is adjusted by controlling the flow rate of propellant with valves located on the fuel lines. The amount of propellant reserved for attitude control is estimated to be about 5% of the total fuel for the mission. This estimate takes into account the longer duration and therefore many more TCM's which will take place compared to previous missions, but also realizes the greater mass of fuel which is being carried for this mission (compared to other missions)[8].

Thermal design of the thruster cluster uses three catalyst bed heaters and valve heaters to maintain the catalyst bed above 200° C prior to firing. The cluster is designed to be thermally isolated from the spacecraft and minimize heat transfer to the cluster or propellant valve to keep the catalyst bed hot. The thrusters are designed to be

Figure 6-A. Thruster Cluster Configuration

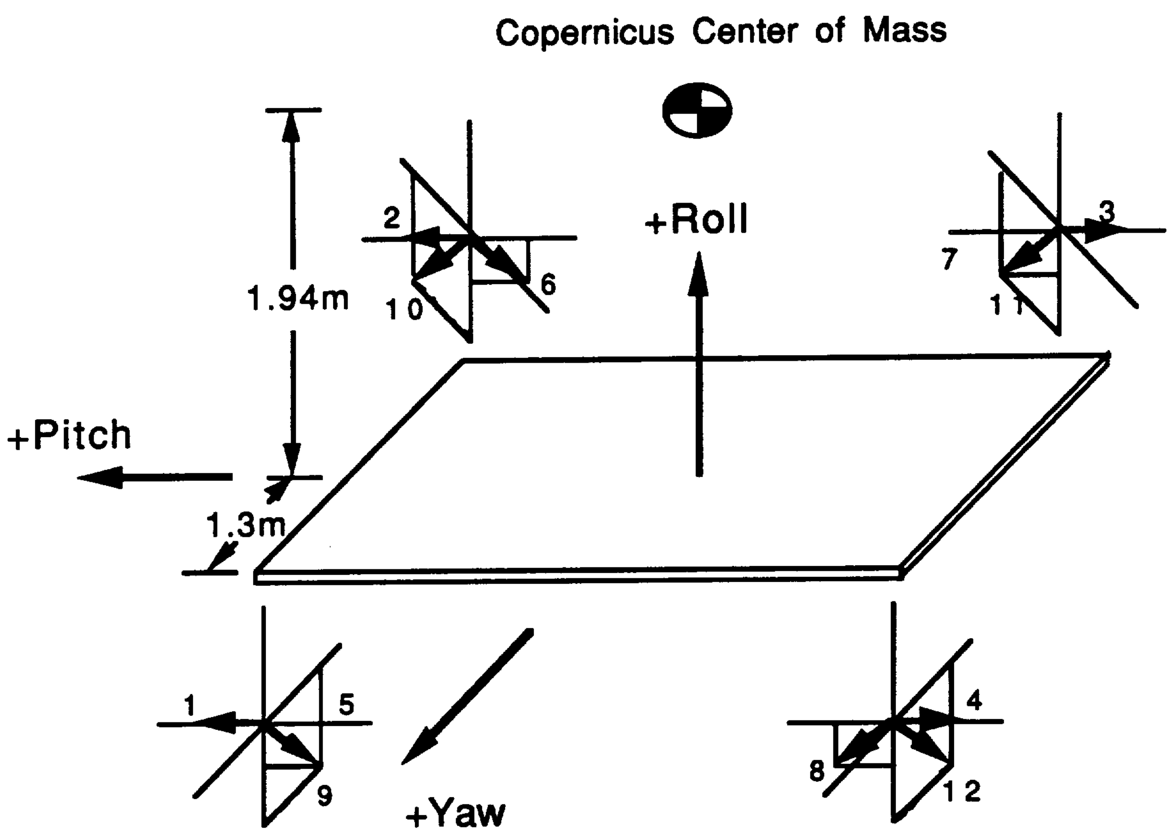

Table 6-E. Thruster Location and Function Matrix

	Thruster Number											
Control Mode	1	2	3	4	5	6	7	8	9	10	11	12
+Yaw					B	B	M	M				
-Yaw					M	M	B	B				
+Roll	M	B	M	B								
-Roll	B	M	B	M								
+Pitch									M	B	B	M
-Pitch									B	M	M	B

capable of achieving mission requirements even in the case of one heater failure.

Algorithms within the main computer control the thrusters to provide three-axis control and to perform closed-loop turns of the spin axis. Such turns may be required up to four times daily to keep the high gain antenna pointed toward Earth, and to orient the spacecraft for TCM's[1].

Conclusion

The Copernicus spacecraft is 3-axis stabilized, using a digital onboard computer, a set of fiber optic gyros, a star tracker, and hydrazine thrusters. Attitude control of the spacecraft is based on measuring spacecraft orientation, estimating spacecraft states, and actuating the thrusters for attitude correction.

The orientation of Copernicus is measured by FORS. The position is calculated using the ASTROS-II. A direct drive actuator with momentum compensation wheel will be used to operate the scan platform. The attitude of the spacecraft will be adjusted with 1N thrusters, located on a structure which surrounds the propellant bladder.

This configuration for the AACS will provide the best control for the long journey the Copernicus will undertake to Pluto.

Appendix 6A. Inertial Control Single Step State State Predictor in Cruise

$$\hat{\underline{x}}_p\ (K/K) = \hat{\underline{x}}_p\ (K/K\text{-}1) + \underline{K}_p[\ Mp(K) - \underline{H}\hat{\underline{x}}_p(K/K\text{-}1)]$$
$$\hat{\underline{x}}_p\ (K+1/K) = \phi(K+1,K)\hat{\underline{x}}_p\ (K/K) + \underline{\Gamma}_p(K+1,K)T_p(K)$$

$$\phi\ (K+1,K) = \begin{pmatrix} 1 & \Delta T & .5\Delta T^2 \\ 0 & 1 & \Delta T \\ 0 & 0 & 1 \end{pmatrix} \qquad \underline{\Gamma}\ p(K+1,K) = \underline{\Gamma}_p = \begin{pmatrix} .5\Delta T^2/J_p \\ \Delta T/J_p \\ 0 \end{pmatrix}$$

The decision to turn the appropriate thruster on at K+1 is based on:

$$E_p(K+1) = (1 \quad K_{rp} \quad 0)\ \hat{\underline{x}}_p(K+1/K)$$

$\hat{\underline{x}}_p(K/K)$ is best estimate of spacecraft pitch statres at K given measurements $M_p(K)$.

$\hat{\underline{x}}_p(K+1/K)$ is the best one-step prediction of S/C pitch state based on $M_p(K)$.

$\underline{K}_p$ is the Kalman gains.

$\underline{T}_p$ is the estimate of torque developed by pitch thrusters.

Process is sequentially repeated in real time.
For yaw and roll axes, the subscripts p are changed to y or r.

Appendix 6B. References

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C-3

CONCLUSION

Conclusion

This proposal for an unmanned mission to Pluto calls for the spacecraft Copernicus to be launched on May 4, 2009 on a 12.6 year journey through the outer Solar System with flybys of Jupiter and Saturn before it reaches its (possible) final destination of Plutoian space.

The proposed design adheres to the previously stated mission requirements and special emphasis was put on optimizing performance, reliability, and mission cost.

This proposal is only a Phase A design report, but it does provide the initial research necessary for later more detailed mission concepts and designs.

Correlation of Primary Design Issues

Primary Design Issue	Related Design Requirements	Options Considered	Rationale for Option Selected
Power Source	Power Requirements Environmental Concerns Interaction with other Subsystems	Only RTG's (MITG)	Length of Mission Modularity increases efficiency over conventional RTG's.
Propulsion	Type of Fuel Used Configuration of Propellant Unit Tank Sizing Interaction with other Subsystems	Electric Propulsion Nuclear Propulsion Solid Propellant Liquid Monoprop. Liquid Biprop.	Flight tested, space storable, simple
Mission Type	Maximize Science Data Minimize Cost	Flyby Orbiter Lander	Attractive delta-V, reliable, simple, low cost
Launch Vehicle	Meets Requirements of Copernicus Reliable, Safety Low Risk	Atlas G-Centaur Titan 34D, Centaur DIT	Satisfies minimum requirements
Orientation Measurement	Reliable, Low Cost Accurate	FORS Laser Gyro Conventional Gyro	Low Cost, Weight Very Accurate Flight Tested By 2000
Star Tracker	Reliable, Low Cost Accurate	ASTROS-II Canopus	Low Cost, Weight, Power Very Accurate Flight Tested By 2000
Scan Actuator	Simple, Reliable Low Cost	Direct Drive Harmonic Drive Two-motor	Simple, Reliable, Low Cost Worked well on Galileo
Attitude Thrusters	Reliable Center of Mass Considerations Low Cost	1N 3N 10N	Length of Mission Simple, Reliable Allows for Redundancy

Correlation of Primary Design Issues (cont.)

Primary Design Issue	Related Design Requirements	Options Considered	Rationale for Option Selected
Material Selection	Micrometeoroid Protection Contamination Protection Radiation Protection	Composites Aluminum Titanium Beryllium	Combination of all to fulfill the variety of requirements.
Thermal Control	Provide Proper Thermal Environ. for Components	Printed Circuit Strip Heaters Multilayer Insulation Blanket Radiating Louvers	Combination of all three for redundancy.
Placement of Components	Scan Platform must have clear view Antenna must be clear COM must be on line of force from main thruster	Nine different combos of components are considered	Ninth configuration was selected because its COM was on the line of force of main thruster. It also fulfilled other requirements.
Antenna Sizing	Proper Communications with Earth	<u>3.7m diameter</u> Other various dia.	Minimum diameter to meet requirements of mission
Accurate Video Imaging	Pointing Accuracy Cost Reliability	Voyager Cameras Galileo Cameras <u>Cassini Cameras</u> New Design	High Tech Data Compressing will be readily available
Type of Science Experiments	Accurate information requested by science	Penetrator/Sample Gathering Atmospheric Investigations <u>Accurate Measur. of planet data</u>	With no previous mission to Pluto it seemed prudent to make general studies and not send unnecessary equipment.

N91-16053 P-90

The Phoenix Pluto Probe
Group 7

AAE 241
University of Illinois
April 24th 1990

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Mission Management

The Phoenix probe which is our design for an unmanned probe to Pluto has an addition which was a driving force to Mission Management. This driving force was the potential use of a Nuclear Electric Propulsion (NEP) system. Though this will increase cost a great deal , it's use has many far reaching effects on the space program. The NEP will not only be at least equal in performance to this mission, but will be shown that in the future it will be cost and performance effective for many missions to come.

Although nuclear power is under the propulsion subsystem, it has such an effect on trajectory and other options that I must study the two, trajectory and propulsion together, to reveal it's true merits for interplanetary travel. The Nuclear Electric Propulsion system has many strong points that lend themselves to the use in such a mission. The strong points for NEP include a continuous supply of power especially away from the sun, low acceleration, and possible trip time savings. These trip time savings are good for long distance mission such as missions past Mars, but are not usable for manned missions. NEP also has a low fuel consumption and high specific impulse, thus making it attractive for missions with a high delta-V, which is definitely a problem when going to Pluto. Another reason NEP is attractive for the Phoenix Probe is the long life time of these reactors, allowing long duration missions with heavy payloads. In fact their was a study done which showed that for more missions expected of a vehicle the cost for NEP decreased. A final point for the use of NEP is that they are safe, increase reliability, and are operationally flexible.

With all these benefits, many of which apply to our probe, we decided to fly an Orbiter mission. The following chart lists the reasons that an orbiter was the best vehicle to fly.

	Flyby	Orbiter	Lander
Scientific:	Minimum Time	Sufficient Time	Maximum Time
Cost:	Inexpensive	Expensive	Very Expensive
Payload:	Light Load	Heavy Load	Heavy Load
Misc:	No Benefits	Future Uses	Unknown surface

As shown on this chart for a Flyby a chemical Propulsion system would be best suited since a Flyby would not utilize a NEP systems strong points. If we consider the distance were going for only one planet with no additional benefits it does not seem to be a wise choice for a mission. For a Lander mission the NEP system works well since it would be a high delta-V mission with a heavy payload, but we don't know anything about the surface so a lander would be a difficult task. We also considered a landers information not equally beneficial for the increased cost, since Pluto is so far away. We decided to Fly an Orbiter mission that would allow our scientific equipment to take more accurate measurements. Measurements with the on board photopolarimeter, solid state Imaging, near Infrared spectrometer, and visible and ultraviolet spectrometer will give us a complete layout of Pluto's thermal properties, landscape, mineralogy, and atmosphere. An Orbiter mission also takes advantage of using the NEP system because it will be a heavy load and an original design, and this new design will be a helpful development for future spacecraft.

The development of a NEP system for our mission is a great advantage for an Orbiter, but there are many missions in the future that would benefit from this technology in cost, time, and performance. In fact many AIAA papers (1,2,3,5,8) think that it is the propulsion system of the future. One mission of the future that would benefit is TAU-a mission to a thousand AU's. This mission is dependent on

NEP if it were to go 1000 AU's in 50 years, to make measurements of the distances to the stars in our own galaxy. A Mars cargo transport mission is also a mission that NEP severely out preforms chemical propulsion in the time to get to Mars and payload carried. Therefore when the Mars initiative begins they would use NEP to send the cargo ahead and have the astronauts rendezvous with it in orbit. A trip back to Neptune using a nuclear propelled Orbiter would take only 10 to 12 years. Using NEP system out performs chemical system when constructing on Orbital Transfer Vehicle(OTV). When this comparison of a NEP OTV vs. a chemical OTV was done it was shown that after initial development, NEP was about $250 million cheaper. This reduced cost over chemical is resulting primarily from reduced propellant consumption and from the larger number of missions which can be accomplished by the single nuclear stage. As shown all these missions plus others are severely benefited by the use of NEP, therefore the sooner it is developed, the sooner it can be implemented to these missions.

The Selection of a launch vehicle for this mission was narrowed down by the fact that our spacecraft weights 24,914 kg. Therefore we could initially eliminate the possibility of using most of todays U.S. launch vehicles, with the exception of using possibly two Titan rockets. We could use two commercial Titans, or Titan 4NUS (Type 1 or Type II). The problem with this would be that we would have to assemble our spacecraft in orbit, which could be done at the space station, but the cost to do all this would be higher than launching it in one launch vehicle not to mention an on-orbit assembly cost.

Another possible launch vehicle would be the Soviet Union's Energia. This launch vehicle is capable of delivering payloads weighing more than 100 tons(90,800 kg.) into a low earth orbit.(6) This payload weight should be sufficient to lift our spacecraft to LEO, plus an upper stage, to lift it into a nuclear safe orbit of approximately 700 km. The obvious difficulty with this is securing the use of Energia from the Soviets. The politics of such an act in itself would be a large accomplishment and if political breakdown occurred then we would be stuck with an expensive spacecraft stranded on the ground.

Other than these two options all the other worlds current launch vehicles can be excluded from evaluation because they would need multiple launches to get our spacecraft in orbit. The cost would be astronomical and on-orbit assembly would be almost impossible, thus satisfying the RFP requirement of minimizing on-orbit assembly. To make our mission at all realistic in a cost and possibility standpoint a requirement is for the U.S. to develop a Heavy Launch Vehicle(HLV). This development is already being considered and planned to satisfy the future needs of NASA.(7) Studies established that a cargo vehicle with increased lift capability (>100,000 lbs.(~45,400 kg.)) would be required by the mid-1990's, to satisfy anticipated civil, commercial, and defense needs.(7) The main goal in these developments is to bring the cost of lifting vehicles to $300/lb of payload delivered to LEO.(7)

The Shuttle-C vehicle can satisfy a variety of missions and meet emerging payload requirements.(7) As currently envisioned the Shuttle-C will be a launch vehicle capable of delivering a minimum of 100,000 lbs. (45,400 kg.) of usable cargo to an altitude of 220 NM (407 Km). The vehicle will be operational in the late 1994 time frame and will incur minimal facility impacts and developmental costs.(7) The Shuttle-C plus an appropriate upper- stage should be able to get our Phoenix probe into a Nuclear Safe orbit(NSO). Therefore the Shuttle-C is the most likely choice for the Phoenix probe and this covers the requirement of identifying the use of a space shuttle.

The final considerable launch vehicle would be the Advanced Launch System(ALS). The objective of the ALS program, being jointly developed by

DoD and NASA, is to define a launch system with a vehicle capable of placing payloads up to 200,000 lbs. (90,800 kg in low earth orbit at a fraction of the cost of today's launch systems.(7) This system design is being cost driven to reduce the total delivery cost to orbit to one-tenth of the anticipated cost for the Titan 4. In addition the launch vehicle must be highly reliable, easily supported and maintained, and responsive to changes in mission requirements.(7) This system has some conflicting information in that some articles say it will be available in the late 1990's while others imply a much longer development time, which I have a feeling is more likely. If this system is in operation at our prescribed launch date it will definitely be the launch system of choice by a cost standpoint.

Out of all of these vehicles the Shuttle-C will probably be our launch vehicle. Shuttle-C is most likely to be ready on time for our mission, cheaper than two vehicles, and easier and more dependable than using Energia, since it will be U.S. made.

To begin in the design of a trajectory I had to first determine what planets would be possible to flyby and thus making the design able to preform several possible missions, an RFP requirement. To determine this I plotted the planets in their approximate positions, at the time that our spacecraft could reach them, with a Earth launch window between 2000-2010.(figure 1) For example Uranus is located where the dark arc is on the circular orbit. The dates on that arc are from 2010 to 2020 assuming an approximate trip time to that distance of ten years. This launch window from 2000-2010 satisfies the RFP requirement. As can be see from this figure, none of the outer planets (Saturn, Uranus,or Neptune) will be aligned with Pluto, therefore these planets are excluded from consideration. Mars and Jupiter are a different story, they will be lined up with Pluto during our launch window. Mars' position is not shown on (figure 1) because it will travel around the sun approximately five and a half times during the launch

APPROXIMATE POSITIONS OF
OUTER PLANETS AT TIME OF
POSSIBLE FLYBY'S WITH
LAUNCH DATE BETWEEN 2000-2010

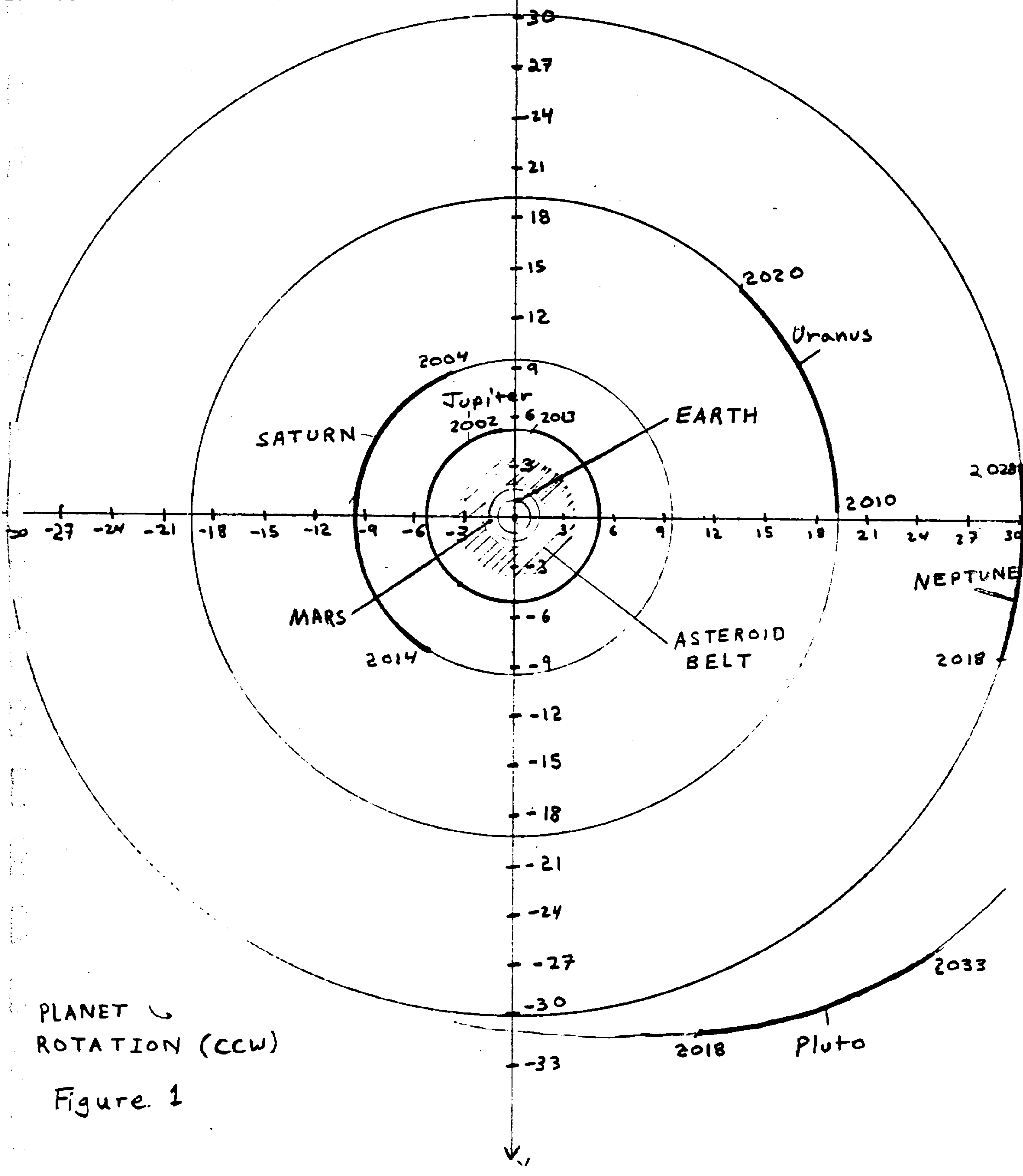

Figure 1

window. A flyby with a gravity assist at Jupiter should give us a tremendous acceleration out to Pluto, so I will try to include this in our trajectory. The other possible flyby's would be Mars and an asteroid. The Mars flyby would be beneficial to help the Mars initiative by searching for a landing site. With reference to the asteroid it is NASA policy that all missions that transverse the asteroid belt should include an asteroid flyby if at all possible, which should not be to hard with 12,000 asteroids out there.

Once I considered what possible missions could be done in addition to our Pluto Orbiter I began our trip to Pluto. First we launch the spacecraft up into Leo and then we use an upper stage, most likely a Centaur, to put the Spacecraft up in a NSO orbit of 700km. At this point we deploy many of the spacecraft booms and scientific equipment. Finally we turn on our Nuclear Electric Propulsion system and our trip begins.

The First part of this trip is to get out of Earth's sphere of influence(SOI). The choice's are to either spiral out of the SOI or to insert into heliocentric space with some booster. The spiral trajectory was chosen because it has a lower mission cost and this spiral out trajectory has direct relevance to future electric propulsion orbit transfer vehicles. The actual spiral trajectory of our Phoenix probe looks very similar to figure 2. The approximation I received using Cheby2 indicates it will take close to 232.3 days to spiral out to escape velocity. During the spiral away from the Earth our spacecraft will revolve around the earth nearly 900 times, thus allowing time for a system checkout. The velocity at NSO will be 7,452 m/s but as the spiral continues it will slow to a final speed of 958 m/s at SOI escape. The last 50 days of this spiral can be seen to be flattening out, this is because the Sun's gravitational influence is becoming stronger than the Earth's. At 925,000 km. from the Earth our Phoenix probe will reach the edge of the Earth's SOI and the origin of the system switches from the Earth to the Sun and our interplanetary trajectory begins.

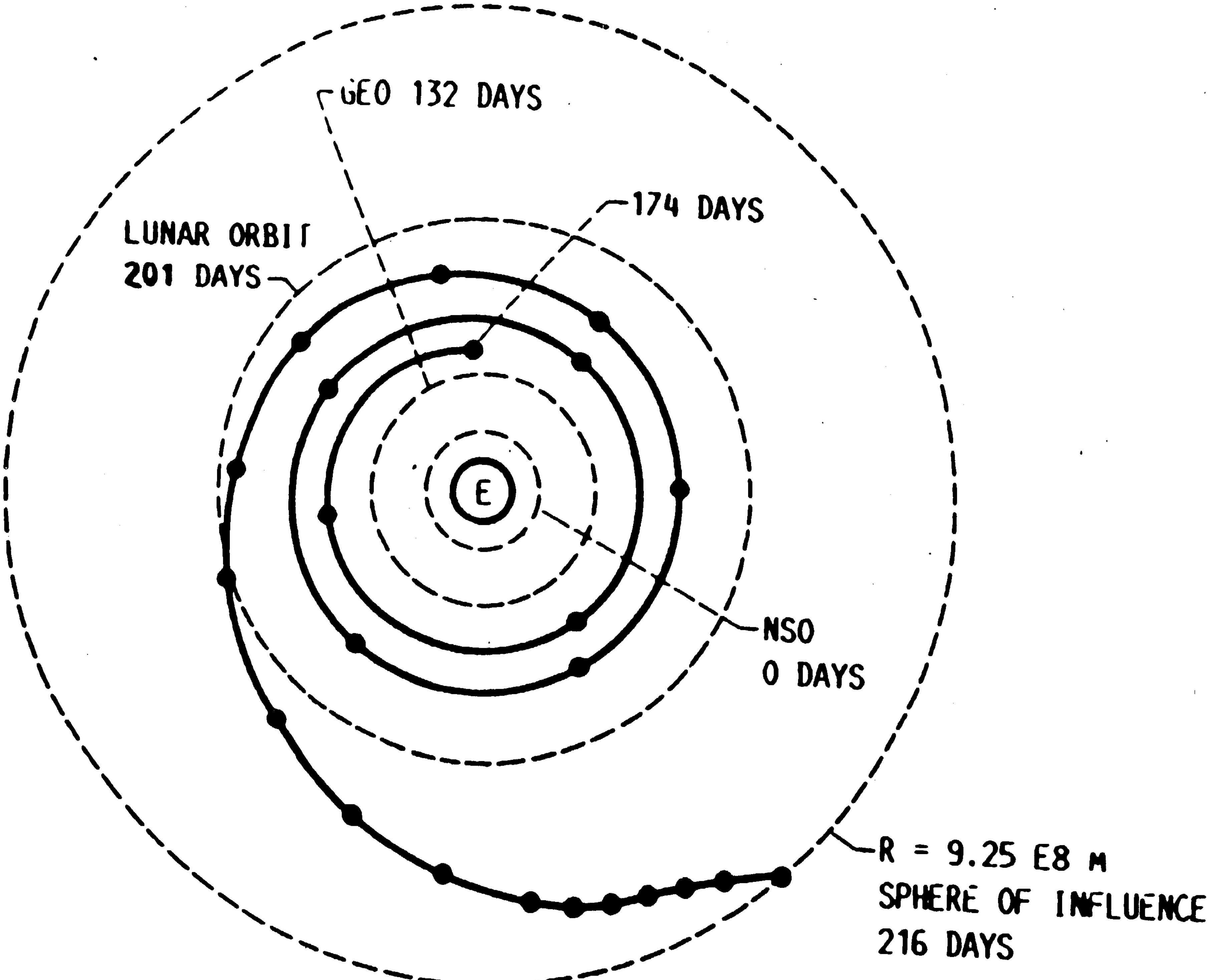

Figure 2 (Reference 1)

All low thrust trajectory analysis was accomplished using the computer code CHEBYTOP2(Chebychev Trajectory Optimization Program). Cheby2 is a multi-purpose trajectory program to optimize either mass or power for low thrust trajectories of either NEP or Solar Electric Propulsion(SEP). I used it to allow simple estimates for variable power from different planets with spiral escapes and spiral capture. The basic information that I used includes; Mass=20,750 kg, Isp=5500, Power=100 kW, Propulsion system specific mass= 57.3, and a power level of 87%. A technical problem that I had was that most of the numbers stated within this paper are at most rough estimates, since this program does not allow for many options and the use of it was limited by the lack of knowledge of its internal working and proper inputs.

The interplanetary travel begins just after leaving Earth's SOI with a solar system speed of close to 30,500 m/s. I ran two scenarios on Cheby2. The first one was a trip from Earth directly to Pluto. The second case prepared consisted of a mission from Earth to Pluto with a swingby at Jupiter. The first case from Earth directly to Pluto included a spiral out of Earth's SOI and a spiral into a elliptical orbit around Pluto. The launch date is to be 2451546 Julian date(JD), Jan. 3,2000, and took approximately 18.5 years. The trajectory when mapped onto galactic map does not look very efficient, this might be caused by the fact that Cheby2 optimizes for power or mass and not for time. This case takes a very long time and is an unlikely choice although our probe could survive that long. This scenario requires the propulsion system to be on for roughly 17.1 years , which our system could handle since it has a lifetime of approximately twenty years. This trip time is again just approximate and with some optimization for time it could be reduced.

The second case of a trip to Pluto with an Jupiter gravity assist came out to be more realistic. The trip time was close to 15 years, with a launch date of 2453095 JD, April 2, 2004, and an arrival date at Pluto of 2458599 JD, April 30, 2019. This trip time of 15 years (5504 days) is more realistic and a better choice over case 1. While analyzing the data for this case I noticed that the trip from Earth to Jupiter, the first 1100 days, seemed very inefficient and has room for improvement. The propulsion system was required to be on for roughly 14 years, thus allowing a great deal of propulsion on time around Pluto. These numbers are just approximations with little or no time optimization.

The reason I stress that these numbers from Cheby2 are approximations is because out of a couple of sources(3,9) information was given for trajectories to Neptune. These missions to Neptune are almost exactly like ours to Pluto, because they use an Orbiter mission, Isp values of 5300 to 5978, and power of 100 kW. The only difference is the fact that they are going to Neptune instead, but in the year that we are planning our mission, Pluto is only 3 to 6 AU's farther away. These papers list trip times of 10-12 years to Neptune, therefore to go an extra couple of AU's shouldn't add more than possibly two years. This indicates a trip time to Pluto of 12-14 years.

A comparison of flight times to get an Orbiter to Pluto using chemical propulsion is just about the same. In fact the best trip time I got with the lowest delta-V was over 15 years also. So there are really no savings in the way of using chemical propulsion, in fact NEP might even get us there faster considering the mass of the Orbiter.

These missions that I planned show no Encounters with Mars nor asteroids. These are not included because Cheby2 does not allow such additions to your flight path. These missions would be very likely to be included although I was unable to determine when the could occur if they could occur. Another obstacle to find an asteroid flyby is to do this there would be a lengthy process of going through 12,000 asteroids and finding those that are near our optimal trajectory.

The orbiting of Pluto is interesting in that on the way there we will have to reverse our thrust vector to begin slow the spacecraft down so that it can enter orbit around Pluto. This reverse thrust should begin to occur 4.6 years before Pluto is reached. Also we will have to do trajectory checks with our sensors to define our position here and along the whole mission to stay aligned with our trajectory. This is very important with a NEP system for we need a longer time to correct trajectory discrepancies. The final insertion into orbit around Pluto will be a spiraling right into an elliptical orbit. With the NEP propulsion system lasting long enough to do all of the scientific studies of Pluto we should be able to raise our orbit and do scientific studies of Charon. The end of our mission will occur when the NEP system finally gives out and we receive no more communications from our Phoenix probe. The two reactors on board should last us up to twenty years and this lifetime is long enough for an adequate safety margin to meet the RFP requirement of being able to carry out our mission plus others. With all this information I have assembled a time line (figure 3) that use case 2.

Costing for our mission is done on figure 4, which itemizes the direct labor, recurring labor hours, and total cost for each subsystem. Our mission cost comes to $4.215 billion to complete whole mission minus the cost of the launch vehicle, which was unattainable since the Shuttle-C is not built yet. This cost estimate includes four spacecraft to be built, thus satisfying the RFP requirements. Although this is an exuberant amount of money you have to weigh this with the new cost efficient subsystem that are being designed, especially the propulsion system. The development of the NEP system is approximately one-third the total cost, so otherwise if this was taken out of our costing the spacecraft would be more cost effective. This price is in disagreement with the RFP, but again one must weigh that against the originality of such a project and it's future benefits.

Costing - figure 4

	A	B	C	D	E
1	Itemized costing				
2					
3		Mass (kg)	Direct L Hours	Recurring L.H.	Total Cost
4	DEVELOPMENT PROJECT		1000 hours	1000 hours	millions $
5	Strucutre & Devices	2758	4796	854	158.7
6	Thermal Control	35	158	49.4	5.4
7	Propulsion	12000	12604	8851	598.4
8	Attitude & Articulation	46.1	739.7	178.1	10.3
9	Telecommunications	62	1831.9	414	23.4
10	Antennas	304	10820	2030	297.6
11	Command &Data Handling	18	215.9	47	6.4
12	RTG Power	5516	7908.1	6123	238.9
13	Landing Radar / Altimeter	50	1303.5	363.7	34
14	Line-Scan Imaging	30	2366.9	636.6	37.4
15	Particle & Field Instruments	49.3	1195.7	483.4	22.6
16	Remote Sensing Instruments	42.1	401.9	32.6	3.7
17	Sampling Instruments	3.5	89.5	15.9	1
18	SUPPORT FUNCTIONS				
19	System Support		7307		237.1
20	Launch +30 days Operations		2388		82.2
21	Imaging Data development		66.5		2.4
22	Sciencce Data Development		150.3		7.6
23	Program Management		2454.4		76.2
24	FLIGHT PROJECT				
25	Flight Operations		10570		358.2
26	Data Analysis		4492.3		160.6
27					
28			Total cost in	1977 dollars=	$2244.2 Million
29					
30			Total cost in	1990 dollars=	$4,214.6 Million

Appendix of Equations

Cheby2 Equations

X+kx/r^3=a where X=Position vector

a=Acceleration vector

k=Gravitational constant of sun

r=|X|

Constant Isp

|a|=(ao /u)(p/po)σ (t) & du/dt=-(ao /c) (p/po) σ (t)

where: ao = Initial acceleration an 1AU

c=Exhaust velocity

u=relative mass of vehicle

σ (t)= 1->powered or 2->coast

Costing Equations

TC=(100%-Z)NRC+RC

NRC=DLC-RC

DLH=DLH(2,M)+(N-2)*(RLH(2,M))/2

where: TC= Total cost

NRC= Non-recurring cost

RC=Recurring cost

DLH=Direct labor hours

RLH=Recurring labor hours

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ACRONYMS

RFP	Request For Proposal
NEP	Nuclear Electric Propulsion
C3	Command, Control, & Communication
STC	Structural and Thermal Control
A&AC	Attitude and Articulation Control
HGA	High Gain Antenna
LGA	Low Gain Antenna
I & C	Instrumentation and control
MAG	Magnetometer
HTR	High Temperature Radiator
JPL	Jet Propulsion Lab

Introduction Structures and Thermal Analysis

The structural analyst in the Phoenix space probe serves three roles; structural design, thermal control and material selection. It is the responsibility of the analyst to make sure that the space probe maintains its integrity for the entire mission. Therefore it will be shown that the Phoenix probe meets its requirements in the *Request For Proposal.* (RFP). Each requirement will be presented along with a description of how this requirement is satisfied. A design configuration will be illustrated along with a description of each component and its interaction with the other components. A mass / inertia configurations will be shown as well as descriptions of launch vehicle compatibility, on - orbit assembly, materials selected, thermal control considerations, and safety issues of Nuclear Electronic Propulsion (NEP). Also, a description of how the structural analyst interacts with the science, propulsion, attitude and articulation control, command, control, and communication (C^3), and mission management will be presented.

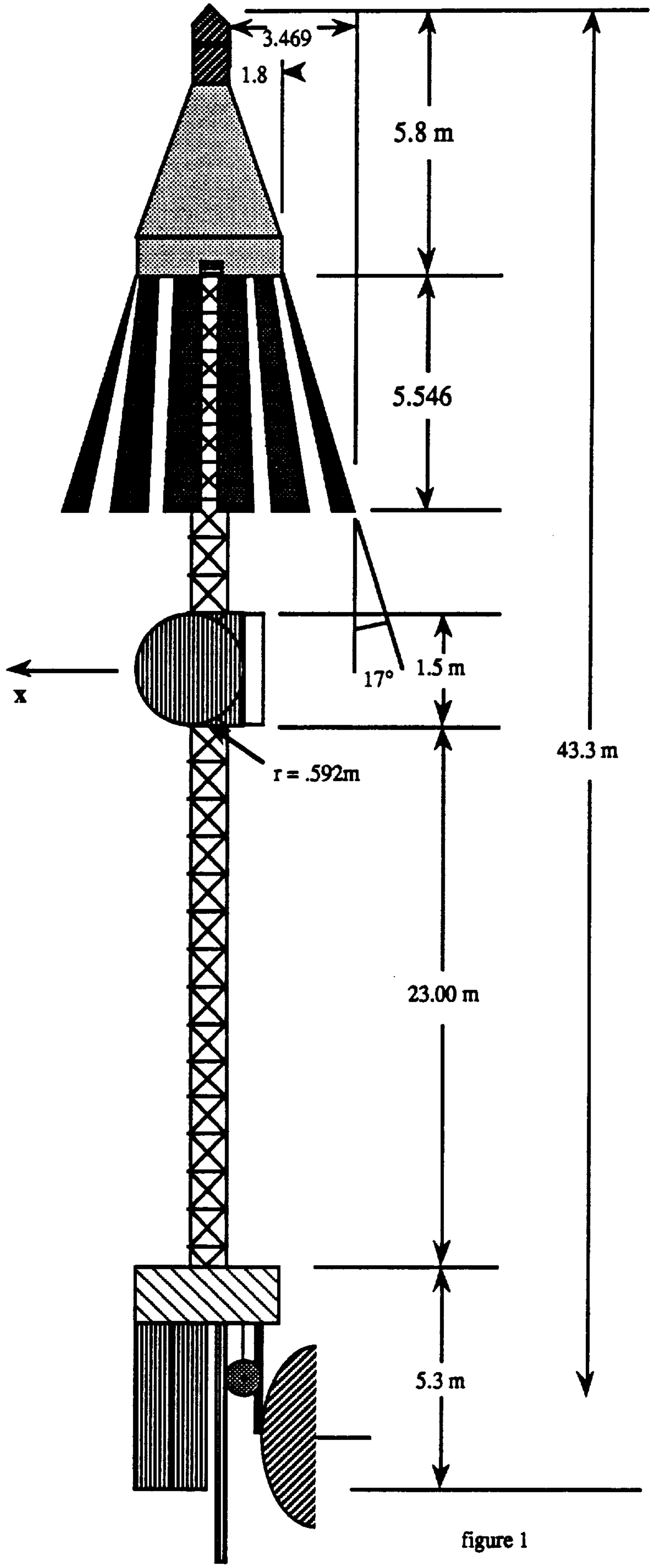
SUBSYSTEM INTERACTIONS

Structures and Thermal Control (STC) is a highly interactive subsystem. STC must work with Mission planning in order to maintain low costing, select a compatible launch vehicle, and most importantly develop a spacecraft configuration that is ideal for a Pluto orbit insertion mission. For the science subsystem STC must provide a clear field of view for the scientific equipment, and maintain equipment at normal operating temperatures. STC provides Attitude Articulation and Control (A&AC) with approximate masses and inertias so that we will maintain stable flight. As with science, STC must maintain C^3 equipment at ideal operation temperatures and provide a clear field of view for the High Gain Antenna (HGA) and Low Gain Antenna (LGA). And finally, Power and Propulsion plays a very important part with STC. The reactors provide 100% of the thermal control for the Phoenix. Also the highly radioactive plume and reactor play a major role in the placement of components.

SYSTEM LAYOUT & DESCRIPTION

Numerous NEP spacecraft configurations have been proposed. Figure 1 illustrates the Phoenix Pluto probe. In this configuration the thrust vector is orthogonal to the vehicle longitudinal axis and the reactor and payload are at opposite ends. The side thrust and end reactor configuration was selected because this design avoids many of the conflicting subsystem requirements that will be discussed later. A clear field of view are provided for the high temperature power system. Thermal control problems are minimized by integrating the spacecraft subsystems along the thermal gradient. [2]

The power module consist of two reactors, a Reactor Instrumentation and Control (I & C) subsystem, shield, heat transport subsystem, power conversion subsystem and the heat rejection panels. The total length of the deployed power module is 11.3 m with the heat rejection panels extending to a diameter of 6.9 m. There are two attitude and articulation thruster units attached the power conversion system directly along the z - plane.



POWER MODULE

2 Reactors
Reactor I & C
Shield
Heat Transport System
Power Conversion
High Temperature Radiator
(HTR) panels
a.k.a. heat rejection panels

PROPULSION

Propellant Tank
6 Main Thrusters

PAYLOAD

Main Platform (AAC housing)
Science & C3 Housing
HGA (4.8m diameter)
LGA
MAG boom (13m)

figure 1

Phoenix
Launch Ready

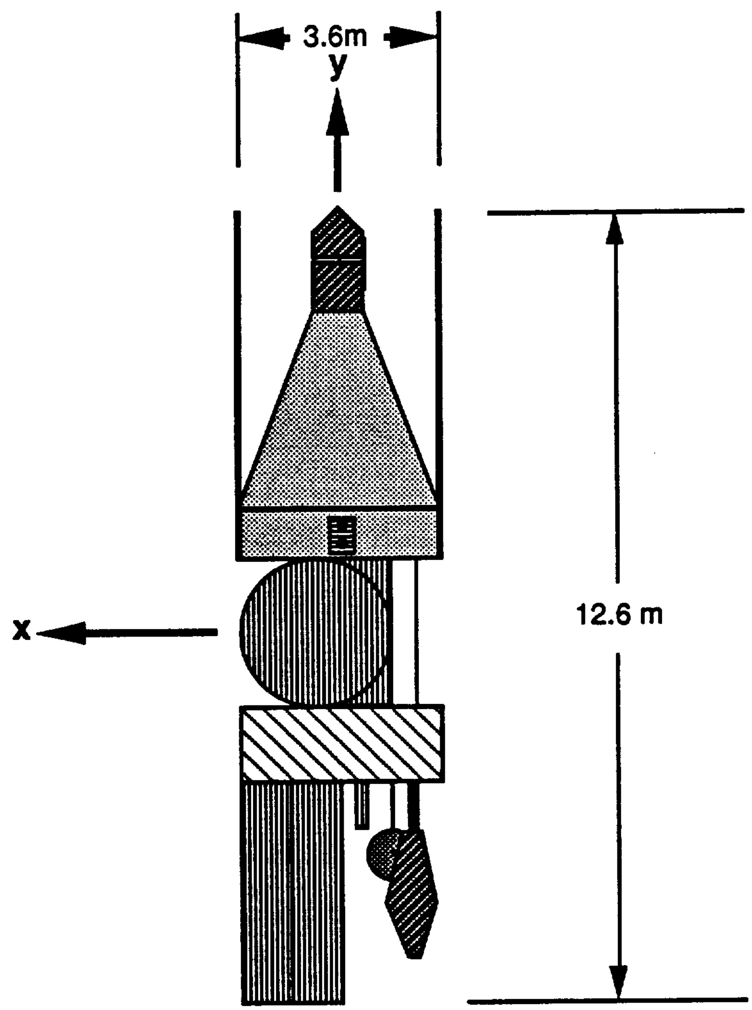


figure 2

The propulsion module is placed on the center of gravity to minimize any unwanted torque due to the thrust. Mercury propellant will be stored in a cylindrical vessel attached directly behind the main thruster unit. The main thruster unit will include the six thrusters needed for our mission. Placed 23 meters down the truss is the payload module. The payload module consist of a main structural platform with a 4.8 m diameter HGA, LGA,Magnetometer (MAG) boom, and a science and communication housing attached. The main platform is designed to house the four reaction wheel assemblies used by A&AC. The science and communication housing features four panels that are kept closed during the majority of the mission in order to protect the equipment from contamination. Once we reach Plutonian orbit and the thrusters are turned off, the science panels are opened allowing a full field of view of Pluto's surface.

Figure 2 shows the Phoenix in takeoff configuration. Notice that the High Temperature Radiator (HTR) panels fold upward. The A&AC thrusters retract in Power Conversion module. The Power and Propulsion boom also retracts into the Power Conversion module and the Payload boom retracts into the payload main platform. On the payload platform the MAG boom retracts and the HGA antenna folds up into its stowed configuration.

Completely stowed, the Phoenix has a length of 12.6 m a diameter of 3.6 m and mass of 20,914 kg (see table 2). The shuttle C is being designed for a 4.57 m diameter, payload length of 25 m, and payload mass of 45,359 kg. Plenty of room and mass is available for packing to insure a safe takeoff.

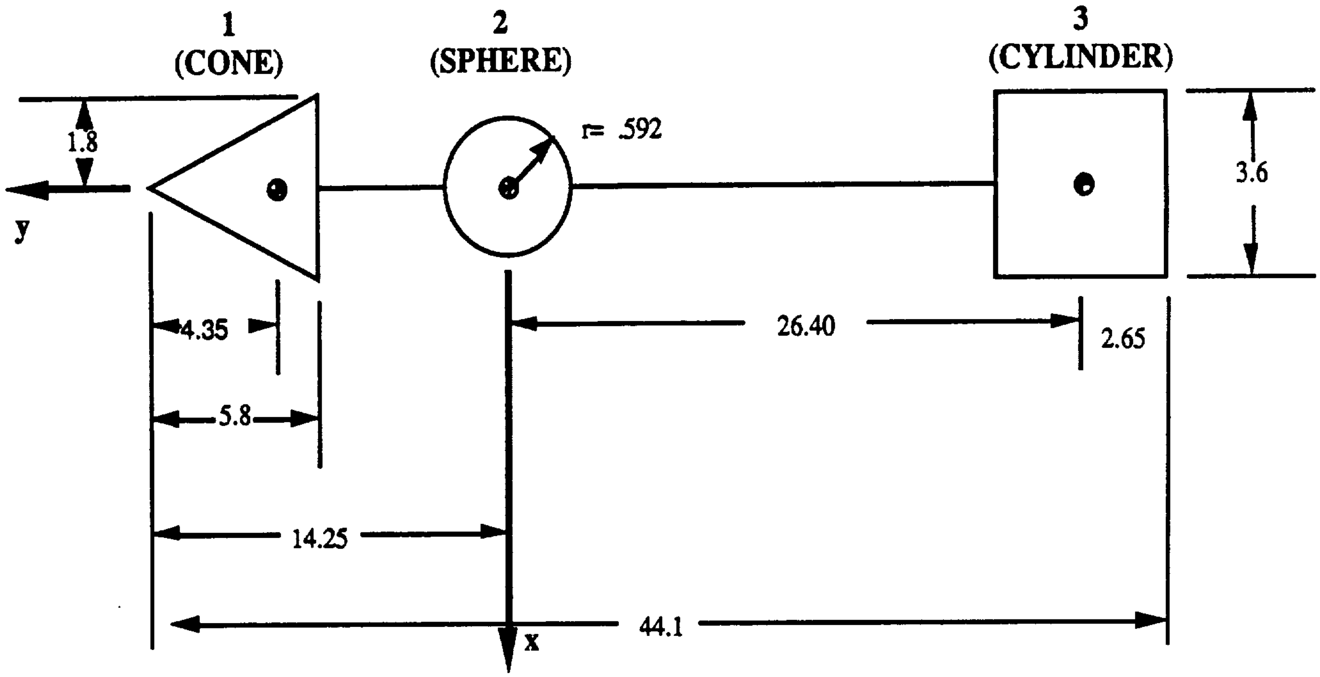
MASS AND INERTIA CONFIGURATION

A summary of the mass breakdown is shown in table 2. A contingency of 20% of the total (dry) system mass is included. The net payload module is 1852.6 kg. An interesting note is that an additional 570 of payload could be added without any additional cost in terms is system interactions. This was calculated with torque and thermal gradient considerations. As shown the net power and propulsion system dry is 5576 kg. But propellant adds an additional 12,000 kg. The subtotal (wet) came out to be 20,914 kg. This mass is only 5.1% different from our initial estimate made during the response to the proposal. Figure 3 shows the simplified diagram of the Phoenix that was used to calculate the mass moment of inertias. The values of these inertias may be found in the appendix.

Table 1 Phoenix Subsystem Mass

ITEM DESCRIPTION	MASS(kg)	
SCIENCE	156.5	
IMAGE SCIENCE SUBSYSTEM		30.0
NEAR INFRARED MAPPING SPECTROMETER		19.5
INFRARED SPECTROMETER		8.2
PHOTOPOLARIMETER RADIOMETER		5.1
EXTREME ULTRAVIOLET		12.3
ULTRAVIOLET		5.2
MAGNETOMETER		5.3
PLASMA WAVE SENSOR		7.2
PLASMA SENSOR		13.2
COSMIC RAY		10.0
DUST DETECTOR		8.5
HEAVY ION COUNTER		4.4
CELESTIAL MECHANICS		10.0
RADIO PROPAGATION		7.6
RADIO MAPPING		10.0
COMMAND CONTROL & COMMUNICATION	350	
S/X BAND ASSEMBLY		4.7
ANTENNA CABLING		3.5
DATA STORAGE SYSTEM		8.6
COMMAND DETECTOR UNIT		10.0
RFS		50.0
HGA (PARABOLOID)		200.0
LGA (HALF-WAVE DIPOLE)		50.0
UNCERTAINTY		23.2
ATTITUDE ARTICULATION & CONTROL	46.1	
TWO AXIS SUN SENSOR (2)		3.2
INERTIAL MEASUREMENT UNIT		15.0
STAR SENSOR ASSEMBLY		4.3
FOUR REACTION WHEEL ASSEMBLIES		25.6
PAYLOAD MODULE STRUCTURE (INCLUDING BOOM)	1300	
POWER & PROPULSION (DRY)	5576	
PRIMARY THRUSTERS (6)		636.0
A A & C THRUSTERS (12)		340.0
REACTOR (2)		1280.0
SHIELD		860.0
HEAT TRANSPORT		445.0
REACTOR I & C		210.0
POWER CONVERSION		315.0
HEAT REJECTION		835.0
POWER CC & D		370.0
STRUCTURE		285.0
SUBTOTAL, LESS CONTINGENCY	7428.6	
CONTINGENCY (20%)	1485.72	
SUBTOTAL PHOENIX (DRY)	8914.32	
PROPELLANT	12000.0	
SUBTOTAL PHOENIX (WET)	20914.32 kg	

PHOENIX MASS/INERTIA CONFIGURATION



(all lengths are in meters) M1= 5,928 kg M2= 12,7632 kg M3= 2223.12 kg

figure 3

Mission Constraints and Requirements

Here is a description of a few of the constraints and requirements given by our project manager and implied by the structural analyst. For a concise listing table 2 illustrates the requirements related to the structural design and summarizes where they are met.

According to the RFP all materials must be available up until 1999. All structural materials for the Phoenix already exist. The support booms are currently flight proven. And the science and communications module will be similar to that of the Galileo and Voyager. But the thermal control of the SP-100 propulsion system has not been thoroughly tested. According to J.F. Mondt of the Jet Propulsion Laboratory (JPL) the generic flight system of the SP-100 will be proven reliable by April 1995. [3]

The use of off-the-shelf hardware is very important in the design of Phoenix. First of all it reduces design and development cost that should be

Table 2 Structural And Thermal Design Requirements

REQUIREMENT	COMPLIANCE
1. Demonstrate understanding of RFP.	Throughout paper.
2. Describe technical approaches used to comply with RFP.	Done in each section.
3. Identify critical problem areas.	Done for each section.
4. Include sensitivity analyses and tradeoff studies.	Done were applicable.
5. Describe method of attack.	MISSION CONSTRAINTS&RQMNTS.
6. Spacecraft must adapt to space environment.	ENVIRONMENTAL HAZARDS & NEP INTERACTIONS
7. Materials used available before 1999.	MISSION CONSTRAINTS&RQMNTS.
8. Identify & minimize on-orbit assembly.	SYSTEM LAYOUT & DESCRIPTION
9. S/C should have sufficient lifetime plus reasonable safety margin.	MISSION CONSTRAINTS&RQMNTS.
10. Stress reliablility, low cost, simplicity.	MISSION CONSTRAINTS&RQMNTS.
11. Weight & cost should be optimized.	MISSION CONSTRAINTS&RQMNTS.
12. S/C should be able to perform several missions.	MISSION CONSTRAINTS&RQMNTS.
13. Off-the-shelf hardware should be used.	MISSION CONSTRAINTS&RQMNTS.
14. S/C should not be a threat to environment or public safety.	SAFETY ISSUES
15. Show & identify layout of components & size.	SYSTEM LAYOUT & DESCRIPTION
16. Verify launch vehicle compatibility.	SYSTEM LAYOUT & DESCRIPTION
17. Give approximate mass & inertias.	MASS/INERTIA CONFIGURATION
18. Describe S/C thermal analysis.	THERMAL ANALYSES
19. Identify materials used.	Done in each section.
20. Show interaction with other subsystems.	SUBSYSTEM INTERACTIONS

directed towards the developing SP-100 propulsion system. The storable HGA, MAG boom assemblies have been featured on the Galileo. Unfortunately, since the Phoenix is such a unique spacecraft, most of the structural components will have to be built for its special configuration. For example, its 30.7 m boom assembly and payload design will be unique. But on the other hand, the materials and methods used to construct these components have been available and flight proven. For example, Carbon fiber/epoxy a light weight, high strength and stiffness material with a tailorable coefficient of thermal expansion and 15 years of proven experience will be used in the boom assembly and support trusses. [5]

ENVIRONMENTAL HAZARDS

The Phoenix Pluto probe has a complicated array of environmental hazards that it will encounter. First is the wide range of temperatures that exits from Earth's atmospheric temperature at take-off to Pluto's orbit that will extend to approximately 34 au for our mission. At these distances the temperature can reach a chilly 42 K. To protect the Phoenix from the effects of such cold temperatures, measures must be taken to keep the all systems within its operating temperatures. These measures will be outlined later in the Thermal Control description.

A second environmental hazard is the meteoroid environment. Large meteoroids are rare in space. Therefore it can be assumed for the purpose of this mission that we do not have to design for this condition. But on the other hand the more numerous smaller meteoroids can present a problem. The effects of these micrometeoroids can be compared to a sandblasting operation [1]. Three systems will be in need of protection; the thin HTR panels, support booms, and the science and communications module. To protect the HTR panels Beryllium Armor will be exposed to the outside surface. To keep the boom assembly from unnecessary exposer it will be enclosed in a single layer Kapton sock. And finally the science module shielding will be roughly equivalent to that of the Galileo spacecraft (0.5 cm aluminum). [2]

A third environmental hazard is radiation. Radiation destroys the orderly structural arrangement of the metals used in spacecraft. Radiation will come from two sources. The first is natural space radiation and the second is the nuclear reactor and exhaust plume. Usually a NEP type spacecraft takes longer to escape earths radiation belts so radiation shielding is important. But in comparison, the Galileo spacecraft was designed for an intense Jovian environment, and the radiation exposer of these two spacecraft are similar. [2] A detailed description of radiation protection can be found below in the NEP Interaction description.

The final environmental hazard is spacecraft charging. As a spacecraft becomes charged, the electrical conductivity can negatively effect the performance of all electronic equipment.

NEP INTERACTIONS

Basically there are two different sources of interaction with the spacecraft by the SP-100 system. Radiation from the nuclear reactor and effects of the propulsion system.

The SP-100 reactor produces both gamma and neutron radiation fluxes. Therefore in order to protect immediate equipment in the HTR, a shield must be present between the two systems. The shield is placed directly behind the reactor and consist of both gamma and a neutron shield. The shield is designed with tungsten as the gamma shield and beryllium as the neutron shield. Lithium-hydride separates the two shield since the materials are not compatible. [2,4]

There are various interactions from the propulsion system that interfere with the spacecraft; 1) surface erosion, 2) film deposition, 3) plasma interactions, and 4) electromagnetic interference. Surfaces exposed to the thruster beam can be eroded. Erosion can cause failure in structural members and thermal control surfaces. The corrosive zone of the exhaust plume is typically 15° but could extend to a 40° maximum. [2] So in order to prevent surface erosion the thrusters point away from all components and the HTR panels will not extend into the 40° cone of the thrusters. The deposition of propellant and non-propellant films on surfaces can cause a serious problem. Propellant and non-propellant sputtered from the thrusters may travel upstream due to diffusion an electromagnetic field effects. These films can alter electrical conductivity and impact antenna performance and thermal properties. [2] The propulsion system is not in danger of these effects because the temperature of these systems is too high to allow these particles to condense on there surfaces. To combat these effects, scientific equipment will be stored in the science and communications module and instruments such as the antenna will be blanketed for protection. The third propulsion interaction is plasma. Plasma generation can cause spacecraft charging and arcing. Circuit logic and breakdown of electrical insulation are results of plasma generation. These problems can be controlled be neutralizing the beam. 2 The final propulsion interaction, electromagnetic interference is produced by permanent magnets and dynamic electromagnetic fields. To prevent such interference, the thruster subsystem should be electrically isolated from other portions of the spacecraft.

SAFETY ISSUES

One of the key requirements of the Phoenix program is safety to Earth's population and environment. The SP-100 has been designed to remain intact and subcritical for a wide range of accident situations, including water immersion, flooding, burial, launch explosions, and reentry.

The unirradiated Uranium 235 fuel does not present a biological hazard. It can be handled and worked around without any special precautions. The reactor will remain unirradiated during ground and launch operations. The shielding around the core prevents the reactor from going critical in the case of water flooding. And the core is honeycombed constructed with absorber rods that protect it from blast or impact. The SP-100 has also been designed with redundant shutdown mechanisms with two independent control systems. To prevent damage during any possible reentry, the nose cone of the reactor is designed with carbon/carbon composites which have demonstrated the ability to increase its strength as the temperature increases. One additional safety feature is that operation of the reactor will not occur until the spacecraft has reached nuclear safe orbit of 925 km. This orbit is high enough that radioactive elements will decay before its eminent reentry.

THERMAL CONTROL

One of the largest problems with the SP-100 is that it dissipates so much heat. For most spacecraft one would be concerned about keeping the various system equipment at a temperature that is warm enough for normal system operation. The SP-100 radiates 2.6 MWt at a radiator temperature of 800 K. heat flux at the radiator is 23,600 W/m^2 which is approximately 17 times the solar heating intensity. To avoid over heating of the science and communication module, at least 21 meters must separate the radiator and the module. (See figure 4). [6] This separation reduces the incident heating on the spacecraft to 1400 W/m^2. To help dissipate the heat into space a system of heatpipes and HTR panels are used. Titanium potassium heat pipes filled with lithium fluid located in the beryllium radiator panels accept heat directly from a source heat pipe assembly. For a detailed description of the heat transport subsystem see fig 5. [3]

CONCLUSION

To sum up, the Phoenix Pluto probe will should prove to satisfy the structural and thermal requirements described in the RFP. The over all

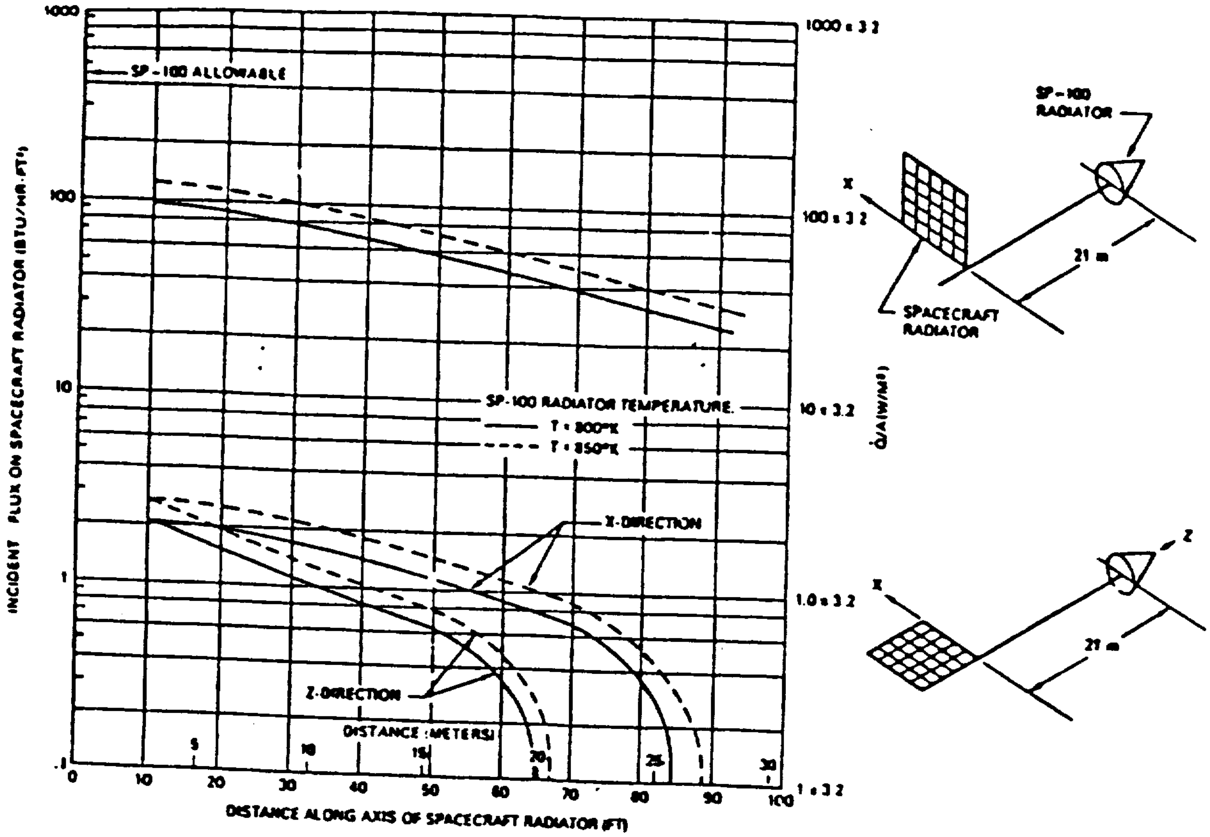


Fig. 4 Spacecraft heating environments from SP-100 radiators.

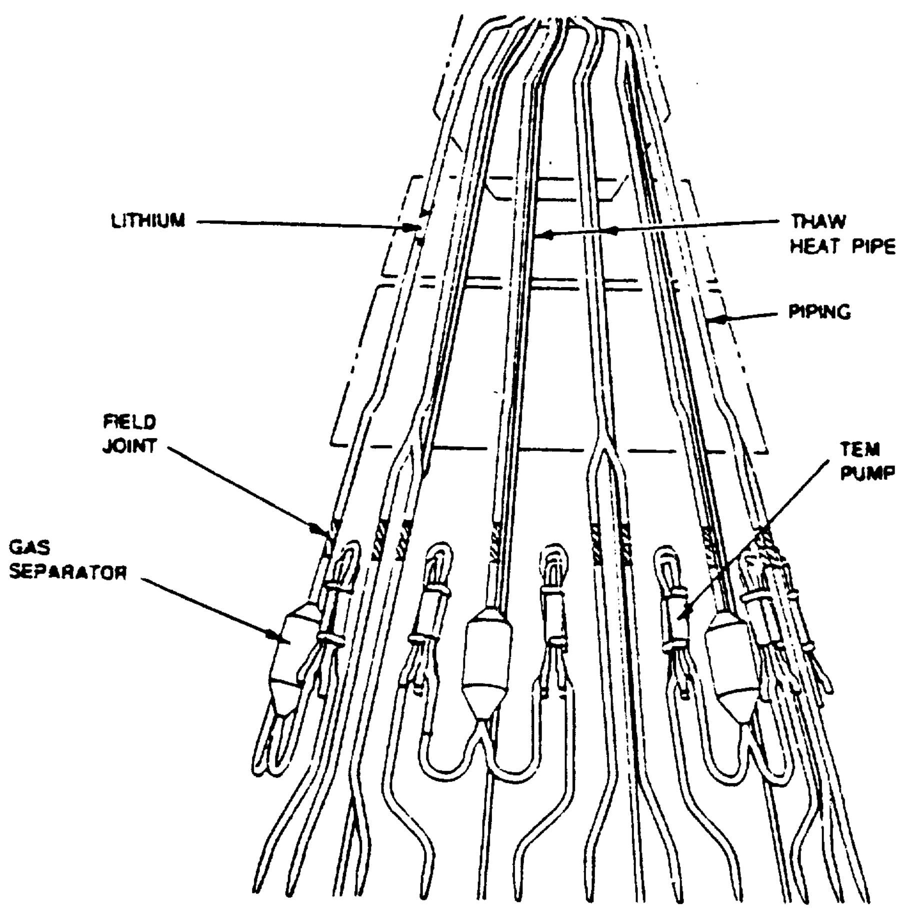


Fig. 5 Heat Transport Subsystem Components

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configuration provides the ideal probe that is required to study Pluto. This is exemplified by the excellent field of view that the science and instrumentation will have. Further more, the SP -100 is the ideal method of thermal control. Not only does it provide ample heat, but also much valuable room on the payload module is saved since all thermal control control comes from the power module.

(16) distance radiation environment

APPENDIX

Equations

Mass Moments of inertia:

CONE

$Iy = 3/10\ (M1)\ (r1)^2 = .3(5928)(1.8)^2 =$ $\underline{1778.4\ kg\ m^2}$

$Ix = Iz = 3/5\ M1\ (1/4\ r1^2 + h^2) + M1\ y^2 = 3/5(5928)[1/4(1.8)^2 + 5.8^2] + (1425)^2 =$

$\underline{1.326E6\ kg\ m^2}$

SPHERE

$Ix = Iy = Iz = 2/5\ M2\ r2^2 = 2/5(16763.2)(.592)^2 =$ $\underline{2349.95\ kg\ m^2}$

CYLINDER

$Iy = 1/2\ M3\ r3^2 = 1/2\ (2223.12)(1.8)^2 =$ $\underline{3601.45\ kg\ m^2}$

$Ix = Iz = 1/12\ M3\ (3r3^2 + L^2) + M3\ y^2 = 1/12\ (2223.12)[3(1.8)^2 + 5.3^2] + 2223.12(26.4) =$

$\underline{65695.05\ kgm^2}$

MASS INERTIA TOTALS

$Iy =$ $\underline{7729.8\ kg\ m^2}$

$Ix = Iz =$ $\underline{1.394E6\ kg\ m^2}$

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Science Instrumentation

1.0 INTRODUCTION

It has been sixty years since a so called planet named Pluto has been discovered, and scientists still do not know exactly what it is. Existing theories state Pluto may well be a planet, but other theories argue that this mysterious entity may be an escaped moon of Neptune or a planetesimal. Basic quantities such as Pluto's albedo, diameter, and density are presently unknown. Scientists believe Pluto to be composed of rock, water-ice, methane-ice, and possibly argon. Charon, Pluto's only known satellite, is even more mysterious than Pluto. Without the knowledge of the above listed measurements, Pluto's and Charon's exact compositions can not be determined.[1] A spacecraft must be sent to the Plutoian system to determine this information. The PHOENIX orbiter, equip with many scientific instruments, is proposed to do so. Although the study of the Plutoian system is the main objective, another goal is to obtain valuable information about Jupiter, Mars, the asteroid belt, and any comet, asteroid, or body the mission may encounter during its planned journey.

Twelve scientific instruments will be used during the course of the Phoenix mission. Four are remote sensing instruments, six are fields and particles instruments, and one is a radio science instrument. The remote sensing instruments are of most importance to the Phoenix mission because they will be able to unlock many of the mysteries the Plutoian system holds. The fields and particles and radio science experiments will correlate information of this type received by previous missions. A detailed description of these instruments instruments are found in part 3.

2.0 REQUEST FOR PROPOSAL(RFP) REQUIREMENTS AND COMPLIANCES

1.) RFP required an unmanned probe to Pluto:

PHOENIX mission complied by developing unmanned spacecraft.

2.) RFP required mission that maximizes information while minimizes cost:

PHOENIX complied by selecting an orbiter with a hope that a needle probe may be developed in time.

3.) RFP requires no materials or techniques after 1999:

PHOENIX Instrumentation Subsystem(PIB) complied by using all instruments with the exception of one which have previously been tested, approved, and used. The one instrument being built of existing technology, of new design, but of no breakthroughs in technology.

4.) RFP required sufficient shelf-life to satisfy mission plus a safety margin:

PHOENIX PIB complies with this demand.

5.) RFP requires mission to be able to perform several missions:

PHOENIX PIB complies with plans to study, Pluto, Charon, and any other planet, asteroid, comet, satellite the path of the mission allows.

3.0 SELECTION, JUSTIFICATION, AND POINTING REQUIREMENTS OF COMPONENTS

3.1 IMAGING SCIENCE SUBSYSTEM (ISS)

OBJECTIVES. The imaging science subsystem is clearly the most valuable scientific experiment carried by the PHOENIX orbiter. Scientists believe Pluto will have a thin or nonexistent atmosphere during the scheduled PHOENIX mission.[1] This will permit an excellent opportunity for an accurate determination of the morphology and geology of Pluto and Charon's surface. The ISS will also map spatial changes in color and albedo, and monitor the variations with time. Other objectives of the ISS will be to locate of the spin axes and rates of rotation of Pluto and Charon. The visual images obtained by the ISS will aid in relating data acquired by other remote sensors to certain features of the plant's surface.[3]

One of the advantages the PHOENIX orbiter offers over a fly-by mission is that the orbiter revolves around the Plutoian system allowing the entire system to be imaged. Also, the orbiter is able to get closer to the system's surface enabling it to take pictures of higher resolution.

When the opportunity arises, the PHOENIX orbiter will study the atmospheres and top cloud formations of other planets such as Jupiter or Mars. Other objects such as asteroids, satellites and comets will also be observed when encountered.[2]

INSTRUMENT. The imaging system used will be the system which is currently being developed for the Cassini misson. The imaging science subsystem consists of a narrow angle camera and a wide angle camera, which share a common set of electronics. The system is based on a 1024 by 1024 pixel charge-coupled device. The ISS is comprised of the following subassemblies:

FILTER WHEEL - This is a two wheel selectable optical filter assembly containing twenty-two filters for the narrow angle camera and fourteen filters for the wide angle camera.

SHUTTER - A dual blade, focal plane, shutter design is used. No preparation is required before exposing an image. The shortest exposure time is five milliseconds. There is no upper limitation.

RADIATOR - Dark current will be subdued by the passive cooling of this radiator.

CCD - The format is 1024 by 1024 pixels, with each pixel size being 12 micrometers square. There are approximately. 50,000 electrons in the partially inverted mode. The UV convertor lumogen phosphor.

OPTICS OF THE NARROW ANGLE CAMERA - The parameters of the narrow angle optics are: Ritchey Chretien with three field correctors; focal length of 2000 millimeters; focal ratio of f/10.5; spectral range of 200-11000 nanometers; resolution per pixel of 6x6 microradians; and field of view of 0.35 degrees square. The close-up lens in the filter wheel begin to fade out of focus at 3.8 km.

OPTICS OF THE WIDE ANGLE CAMERA - The parameters of the wide angle optics are: refractor in type; focal length of 250 millimeters; focal ratio of f/4.0; spectral range of 350-1100 nanometers; resolution per pixel 48x48 micro radians; and field of view of 2.8 degrees square.

Other subassemblies which will not be described here are: the detector head, square root processor, image data compressor, director and signal chain logic, and power supplies. For more information on these subassemblies see reference 4.

The ISS described described above is of new design, but, will be of existing technology. If this design is not perfected by the time of the mission, the imaging system used on the Voyager mission shall be used instead.

The narrow angle camera, wide angle camera, and common electronics module will be mounted on the scan platform and inter-connected by shielded cables.[4]

3.2 NEAR - INFRARED MAPPING SPECTROMETER(NIMS)

OBJECTIVE. The main objective of this experiment is to investigate the near-infrared spectrum to determine the geology of Pluto and Charon. The experiment will also map and determine the mineral content of the surfaces of these bodies.

Pluto is believed to be composed of methane-ice, water-ice, and possibly argon, neon, and nitrogen. These molecules along with others will be specifically monitored by the NIMS. Other objectives of this experiment will be to probe the atmospheres and cloud layerings of Jupiter, Saturn, Mars, and any other objects with atmospheres when the opportunities arise.

INSTRUMENT. The NIMS was selected because it combines imaging and spectroscopic abilities in the same instrument. The telescope subassembly consists of an all- refractive telescope with a 22.9 cm aperture Ritchey Chretien. The focal length is 800 mm with an aperture of f/3.5.

The spectrometer subassembly consists of: a Dall-Kirkham type of collimator, a wide angle, flat field camera, and plane grating. The collimator has a focal length of 400 mm and a ratio of f/3.5. The camera's focal length is 200 mm, with a f/1.75 focal ratio. The grating is dual blazed, with 400 lines per mm.

The detectors (fifteen) are of the most sensitive type available, indium antimonide. They require cooling by a passive radiator to 80 K. Each of the 15 detectors is placed in different areas to sample specific regions of the spectrum. The NIMS is designed to measure wavelengths in the range of 0.7 to 5 micrometers.

The NIMS consumes an average of 8 w, and weighs 18 kg. The Galileo carried a NIMS of the above type. The NIMS will be positioned on the scan platform near the ISS. For more information on this instrument see reference 2.

3.3 PHOTOPOLARIMETER - RADIOMETER (PPR)

OBJECTIVE. The primary objective of the PPR experiment is to measures the polarization and intensity in the region of visible light (400-700 angstroms). This data will yield information about the properties of light-scattering surfaces.[3]

A second objective will be to measure the thermal radiation of Pluto and Charon. Another objective is to find the radiation budget of the Plutoian system by measuring the total thermal emission and reflective solar radiation.[2] The above stated objectives will also be applied to the atmospheres of Jupiter and any other planet with an atmosphere when encountered.

INSTRUMENT. The PPR used on the Galileo mission was the instrument selected to be carried by the PHOENIX mission. It was selected because of its dual abilities to measure photometry and infrared radiometry. The instrument is equipped with a Dall-Kirkham telescope with 10 cm aperture and a 50 cm focal point. This is the primary optical path of the subsystem. This optical path collects light and passes it through selected filters. This collected light is then measured by detectors.

There are two minor optical paths in the PPR. The first of these paths gathers radiation from the surveyed object. The other minor path collects radiation from space. These minor optical paths are used only in the radiometry mode of the instrument. Infrared channels in the radiometry mode are set below 4 micrometers, at 17, 21, 27.5 and 37 micrometer, and above 42 micrometers.

In the photopolarimerty mode, only radiation entering the primary optical path is emitted to the detectors. A beam is passed through a filter and enters in to a Wollaston prism. By rotating the filter wheel, the polarization of the transmitted beam rotated 90 degrees. This determine the orientation of the polarization of the incident beam. Polarimetry channels are centered at 4100, 6780, and 9450 angstroms. Photometry channels are centered at six positions between 6180-8920 angstroms.

The PPR subsystem has three important safety features: deployable covers which shield all optical when thrusters are fired, sunshades which prevent sunlight from directly entering, and replacement heaters which maintain the temperature when the power is turned off. The PPR subsystem weighs 4.8 kg, uses a peak power of 10 watts, and is mounted on the scan platform with the other remote sensing instruments.[2]

3.4 ULTRAVIOLET SPECTROMETER(UVS)

<u>OBJECTIVE.</u> The main objective of this experiment is to determine the structure and composition of the atmospheres of Pluto (if there is one), Charon, and any other satellite of Pluto which may exist. Atmospheric gases discharge radiation at ultraviolet wavelengths for two reasons. They are sometimes excited by bombardment with energetic particles, and sometimes the resonance dispersion of solar ultraviolet radiation cause this.[3]

Airglow will be analyzed by the UVS. The UVS will also determine ultraviolet reflective properties of the surfaces of these bodies. This will yield information to help characterize surface materials and their physical state.[2]

INSTRUMENT. The PHOENIX mission selected an ultraviolet spectrometer similar to the instrument carried by Galileo. This instrument consists of a Cassegrain telescope (250 mm aperture), a monochromater, three detectors (photomultipliers), and control logic. The telescope is unique in that it can sample ultraviolet radiation coming from a small portion of the atmosphere or surface. The field of view produced by the spectrometer is 0.1 by 1.4 degrees for 1100-1900 and 2800-4300 angstrom detectors and 0.1 by 0.4 degrees for the 1600-3000 angstrom detector. The monochromator has a focal length of 125 mm.

A programmable grating drive which is regulated by the control logic controls the wavelength of the radiation being measured. The grating supplies a resolution of 13 angstroms in the first order spectrum and 7 angstroms in the second order spectrum. The photomultipliers are capable of investigating wavelengths from 1150-4300 angstroms. Photon pulses are counted every 0.0007 seconds. This UVS was selected because of its wide range of spectra (1150-4300 A^{o}) and its flexibility in variety of data taking programs.[2]

The UVS subsystem weighs 5.21 kg, and consumes 5.33 W at 2.4 kHz and 50 Vac. It is secured on the scan platform with the previous three instruments.[2]

* NOTE: No direct sunlight can enter any of the remote sensing instruments. All instruments shall be equip with shields to block the sun.

3.5 MAGNETOMETER SUBSYSTEM(MAS)

OBJECTIVE. Interplanetary space is traveled by the solar wind, streams of charged particles, and shifting magnetic fields that the solar winds bring with them. Some planets have their own magnetic fields. The main objective of this experiment is to determine if Pluto and Charon possess magnetic fields. The second objective is to investigate interactions between Pluto's and Charon's magnetospheres, if any exist.

The magnetometer experiment will also acquire data on all other magnetic fields encountered during th e Phoenix mission. This data will be used in comparative studies with data received from other fields and particles instruments.

INSTRUMENT. The magnetometer subsystem consists of four subassemblies; two high field magnetometers (HFM), which measure $\pm$0.5G to $\pm$20G, and two low field magnetometers (LFM), which measure $\pm$8.8 gamma to $\pm$50,000 gamma. The Phoenix orbiter does not spin, therefore the type of magnetometer that was carried on the Voyager mission will be used.

Each of the four subassemblies consist of triaxle fluxgate magnetometers that measure field and intensity along three orthogonal axes simultaneously; thus, producing direct vector measurements. One LFM is placed at the middle of the boom (0.80 kg), and the other is placed at the end(0.75 kg). This arrangement will allow the spacecraft's magnetic field to be separated from the ambient magnetic field. In doing this, accurate information can be obtained. Both HFMs are placed near eachother, at the proximal end of the boom (0.26 kg each). The total mass of the MAS is 5.72 kg.[3]

3.6 ENERGETIC PARTICLES DETECTOR(EPD)

OBJECTIVE. The main objective of this experiment is to investigate the temporal fluctuations and spatial disbursement of ions and electrons in the medium to high energy range (0.015 to 0.2 MeV and 0.1 to 1.0 MeV respectively). This experiment will be performed in the Plutoian system, interplanetary space, and other systems when encountered.

INSTRUMENT. The EPD has two bidirectional detector telescopes which are mounted on a platform in the spun instrument section. The telescopes used are a low-energy magnetosphere measuring system (LEMMS) and a composition measuring system (CMS). The LEMMS includes an ion telescope, two detectors, and a magnetic electron spectrometer. The energies measured by this subassembly are .015 to 0.2 MeV and 0.1 to 1.0 MeV. The CMS is comprised of a three-parameter detector system consisting of nine detectors. These detectors measure the energy spectra, composition, and pitch angle distributions of energetic ions in the Plutoian system. The EPD subsystem has a total mass of 10.77 kg and is located on the spun instrument section.[2]

3.7 PLASMA SUBSYSTEM(PLS)

OBJECTIVE. Plasma is gas found in space that is electrically neutral, but, composed of charged particles. The main objective of the PLS experiment is to measure plasmas velocity, density, and pressure. PLS instrument also determines the plasma flow direction by measuring the variation velocity with direction.

INSTRUMENT. The PLS subsystem used on Galileo was selected over the PLS subsystem used on the Voyager for the following reasons. First, it has an extended energy range of 1.2-50,400v; where as the Voyager PLS had a range of 10-5920v. Second, it has three miniature mass spectrometers which analyze ion compositions, while Voyager had none. Finally, while Voyager's PLS had a temporal resolution of 100 seconds, Galileo's PLS has a temporal resolution of 5 seconds.[2,3]

3.8 PLASMA WAVE SENSOR(PWS)

OBJECTIVE. The objective of this instrument is to identify and analyze the radio and plasma waves in Pluto's magnetosphere. The PWS is equipt with the the capability of remote sensing of source location. Magnetospheres of other planets and satellites will be studied when opportunities arise.

INSTRUMENT. The PWS consists of an electric dipole antenna for the detection of electric fields and two coil magnetic antennas for the detection of magnetic fields. These subassemblies measure spectral characteristics of electric and magnetic fields in the range of 5 Hz to 5.65 MHz. The total mass of the PWS is 7.22 kg. The antennas are located at the end of the magnetometer boom on the vertical axis. [3]

3.9 DUST DETECTOR SUBSYSTEM(DDS)

OBJECTIVE. The dust detector experiment will aid in the understanding of physical and dynamic properties of small dust particle in the Plutoian system. This information will help answer questions about the existence of Charon, which is thought by some to be a fragmented piece of Pluto.

INSTRUMENT. The DDS is comprised of a set of grids that sense the impacts of dust particles. The instruments field of view is 140 degrees. It can measure masses in the range of 10^{-19} to 10^{-9} kg and velocities in the range of 2 to 50 km. The DDS measures 0.1 by 0.1 m, weighs 4.37 kg and is placed on the spun instrument section to determine the flight direction of the particles.[2]

3.10 MICROMETEOROID DETECTOR(MMD)

OBJECTIVE. Micrometeoroids are particles smaller than one mm in diameter that are present in the space occupied by our solar system. Although the Voyager mission took no particular notice to the asteroid belt due to the results of the Pioneer 10 and 11(no concentration within the belt), the Phoenix mission will carry a micrometeoroid detector (MMD) to study the belt and verify Pioneer's findings.

A second reason for employing this instrument is to study the Plutoian region for these particles. A knowledge of the micrometeoroids present in this area may unlock some of the mystery of the being of Charon. It may give some clues as to if Charon is a fragmented piece of Pluto.

INSTRUMENT. The MMD used on the Phoenix mission is similar to the instrument used on the Mariner-Mars spacecraft. A crystal acoustical transducer is fastened to aluminum plates (22 cm by 22 cm). The crystal will discharge an electrical pulse whenever a micrometeoroid strikes the plate. The plate is completely covered with an insulting and conducting film. This forms a capacitor sort of detector. A potential is placed across this capacitor and an electrical discharge occurs when a micrometeoroid perforates the insulation of capacitor. This type of capacitor detector is self repairing and is excellent for repeated use. When the capacitor detector output coincides with the output of the acoustic detector, the direction of the micrometeoroid can be determined. The present design of the MMD allows for the determination of the number and penetration power of the micrometeoroids.

New MMDs which will calculate velocity as well as momentum may be available before the Phoenix is built. This advanced instrument will be used in place of the above described MMD if so.[5]

3.11 RADIO SCIENCE SUBSYSTEM(RSS)

OBJECTIVE. Two experiments, celestial mechanics and radio propagation, will be investigated by the radio subsystem. The celestial mechanics experiment will be used to determine the structures and shapes of the gravitational fields of Pluto and Charon. This subsystem uses the radio system to perceive gravitational perturbations on its trajectory.

A primary goal of the radio propagation experiment is to study ionospheres, atmospheres, and magnetospheres. This will provide measurements of density, pressure, and temperature as a function of height; which is dependent on the doppler shift. While not as important for the probing of Pluto, the experiment will be more essential for the studies of planets with atmospheres.

INSTRUMENT. The radio frequency subsystem is used in combination with receivers and transmitters based on earth. The RFS measures doppler shifts, echo time delays, amplitude, spectrum and polarization of radio signals. The mass, size, and location of this assembly can be located in the Command, Control, and Communications subsystem. [2]

4.0 SCIENCE TIMELINE

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1 2 3 4 5 6 7

EARTH PLUTO

1 - Instruments conduct observations of earth and Moon to calibrate

2 - Cruise mode

3 - Fields and particles instruments begin operating continuously

4 - Study asteroid belt (2.2 to 3.5 AU)

5 - Perform trajectory maneuvers to cancel launch injection errors and refine aiming

6 - Remote sensing of Plutoian system

7 - Scan and photograph star field

* - Every 0.25 AU scanning instruments will perform remote sensing

* - Observe all planets, satellites, meteors, and comets trajectory passes

References

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2.) National Aeronautics And Space Administration, Washington D. C.; Galileo: Exploration of Jupiter's System; NASA SP 479, pg. 119-141; 1985.

3.) NASA Jet Propulsion Laboratory, California Institute of Technology; "Voyager Neptune / Interstellar Mission: Flight Science Office Science and Mission Systems Handbook"; JPL D-498, PD 618-128, pg. 1-52; October 1, 1987.

4.) NASA Jet Propulsion Laboratory, California Institute of Technology; "Cassini Mission: Saturn Orbiter Proposal Information Package, Volume VI: Imaging Science Subsystem Description"; JPL D-6464, PD 699-11, Vol. VI; October 10, 1989.

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ATTITUDE AND ARTICULATION CONTROL SYSTEM

The goals of the attitude and articulation control system (AACS) are to achieve and maintain a particular orientation in space. The first phase of this process is attitude acquisition which employs a variety of sensors to locate the spacecraft in space relative to some inertial reference frame. Stabilization of the craft in this orientation is maintained through the use of control actuators which must also be capable of maneuvering the spacecraft from one attitude to another.[4] The selection of the AACS methods and hardware depend on the mission requirements, with special care taken to insure compatibility and integration with the other subsystems.

AACS REQUIREMENTS

Table 1 outlines the specified and derived requirements pertaining to the AACS , and provides a reference location of compliance for each requirement. The primary requirements of the AACS are to survive the long life of the mission and be capable of several different missions. The first of these leads to the derived requirement of total redundancy of all systems, while the mission flexibility requirement calls for a reliable system of control actuation. Also the fifteen-plus year life of the mission dictates the need for autonomous control. An increasing communication delay time as the spacecraft moves further away from earth and periods of no communications require an on-board system capable of analyzing attitude acquisition information and implementing control actuation to maintain spacecraft stabilization without the benefit of command. This is accomplished with advanced software on-board with preprogrammed actuation sequences to accommodate all

AACS REQUIREMENTS

Table 1

SPECIFIED REQUIREMENTS (R.F.P)

1. Optimize performance, weight, and costs in design trades.
2. Design must be reliable, low cost, simple, and easy to operate.
3. Use "off the shelf" hardware developed before 1999, when available.
5. System should have a sufficeint lifetime plus a safety margin.
6. Must be an original and imaginative design.
7. Identify the design approach and technical probl ems.
8. Probe must be capable of several missions.

DERIVED REQUIREMENTS

1. Maintain antenna and science instrument pointing
2. Select a stabilization method.
3. Select types and placement of sensors and actuators.
4. Integrate the AACS with other subsystems.
5. Determine torque and momentum requirements.
6. Must have partial autonomous control capability.
7. Determine environmental effects.
8. Must have a fifteen year minimum lifetime.
9. Total redundancy of all systems

conceivable maneuvering scenarios.

Further requirements of the AACS are dictated by the basic structural configuration of the flight vehicle. For instance the dumbbell type configuration selected for the final design must be Three-axis stabilized. Spinning the vehicle about the pitch axis (x-axis) or the thrust vector (z-axis) would result in poor communication capability since the antenna must be placed at the far end of the spacecraft to avoid adverse interaction with the nuclear propulsion system. Spinning about the roll axis (y-axis) would result in an unstable spin which would eventually lead to an undesirable end over end rotation about the pitch axis (x-axis). All other requirements are dependent upon AACS component selection and placement and are discussed throughout the report.

DESIGN APPROACH

The method of attack for selecting the AACS is basically a design by design approach. Following a considerable amount of initial research, several spacecraft and AACS configurations are selected with input from the other subsystem analysts. These preliminary design choices are then analyzed to determine if they satisfy the real and implied requirements of the mission. All problems with the selected systems are then outlined and further research is done to determine possible solutions to these problems. Finally the options are compared and a final configuration is selected. The remaining analysis consists of refining the best choice and presenting the final design.

DESIGN TRADES

The first design trades considered are low cost versus reliability, long life, and accuracy. This cost pertains to both weight and monetary cost and is a factor in the selection of the AACS hardware. Another important trade related to hardware selection is an original design versus "off the shelf" hardware. Newer components may be technically superior but previously space tested hardware has the overwhelming advantage of known performance parameters, which reflects the use of tested components in the final design configuration. Other trades relative to the final design include maneuverability versus disturbance sensitivity and reaction wheel versus thruster control in terms of stabilization capability and fuel consumption.

INITIAL CONFIGURATIONS

Three different spacecraft and AACS configurations were selected for the preliminary design analysis.

They include:

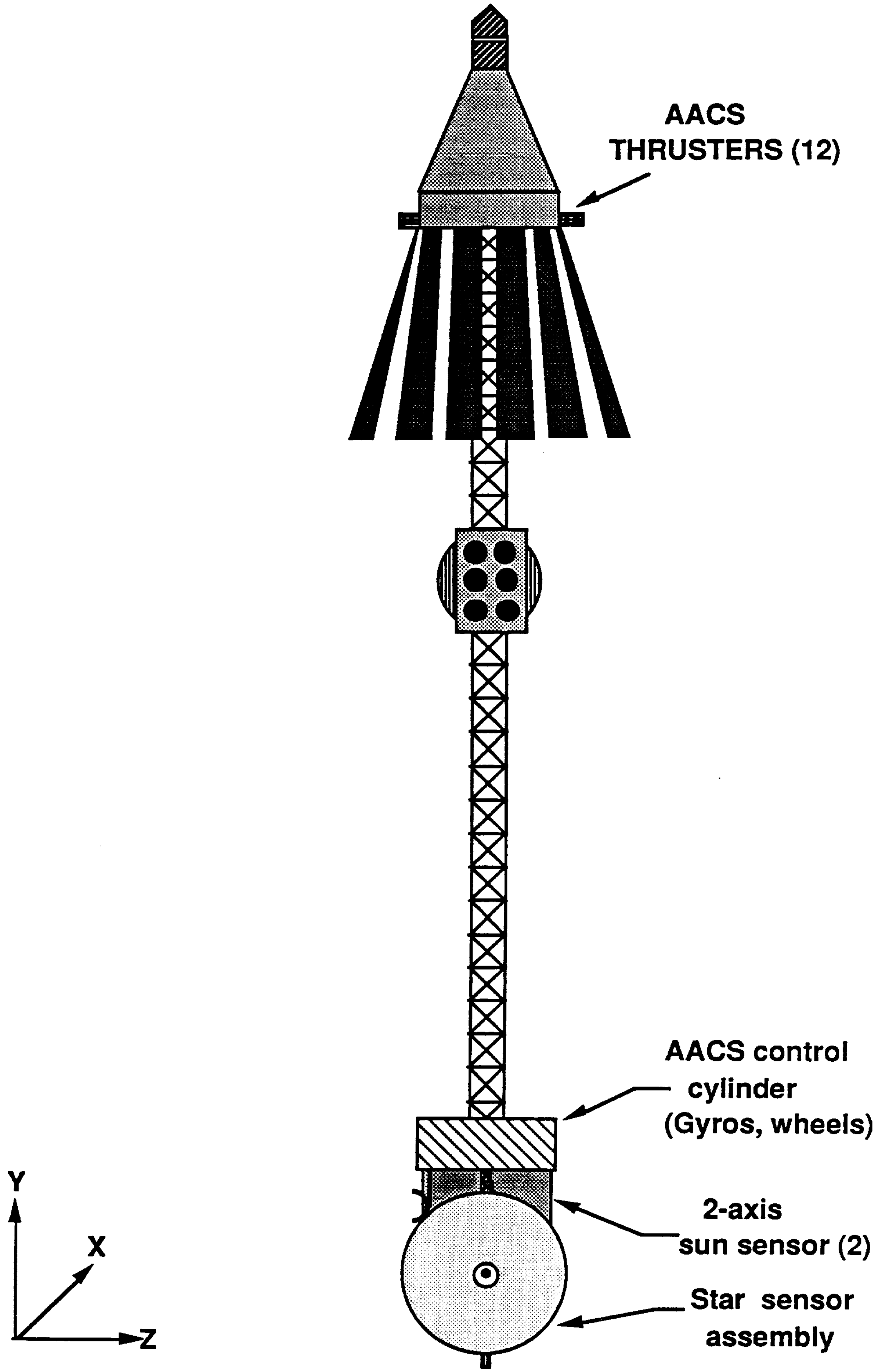
1. Spin stabilized spacecraft - Chemical propulsion, RTG power.
2. Spin stabilized spacecraft - Nuclear electric powered upper stage.
3. Three-axis stabilized spacecraft - Nuclear electric propulsion, two on-board reactors.

The first choice is a Pioneer type scientific probe with hardware modifications made to fulfill the mission requirements, such as long life. This configuration was rejected without further research due to its incompatibility with the nuclear electric propulsion (NEP) system selected by the the design team.

The second spacecraft configuration utilizes Three-axis stabilization throughout the initial thrust phase of the mission, which is limited by the assumed ten year life of the NEP system. At this time the entire NEP system is jettisoned and a spin stabilized scientific probe continues on to Pluto powered by RTG's. The advantage of this particular configuration is that the NEP upper stage can deliver a larger payload Through the initial delta-v required than a weight comparable chemical upper stage.[1] Also, following the NEP system detachment, the scientific probe would only require a five year power active lifetime, assuming a fifteen year mission. Disadvantages of this selection include a large launch mass and a loss in simplicity of design. This configuration would would require two independent control systems, one for the three-axis control of the primary vehicle and another for the spin stabilized craft. Also a large change in the mass of the vehicle following the NEP system detachment would require a complex control scheme to maintain stability. These drawbacks and the resulting high monetary cost of such a mission do not satisfy the specified mission requirements.

FINAL DESIGN CONFIGURATION

The third preliminary configuration was selected as the final design on the basis of mission requirement compatibility and a favorable analysis of the design trades. A layout of the spacecraft including locations of the AACS components is shown in figure 1. The vehicle consists of two nuclear reactors, a power conditioning unit, and heat shielding at one end, and the scientific payload and C^3 hardware at the opposite end. The spherical fuel tank is located directly below the main thruster block, both of which are positioned at the vehicle center of mass. As discussed earlier, three-axis stabilization is the only viable control method for this dumbbell type configuration due to the requirements of maintaining adequate communication capability while avoiding adverse interaction with the NEP system. Furthermore, a flexible system utilizing active control is desirable to counteract the effects of structural vibrations within the 28.5 meter extendible boom.[5]


AACS
THRUSTERS (12)
AACS control
cylinder
(Gyros, wheels)
2-axis
sun sensor (2)
Star sensor
assembly
Y
X
Z

The three-axis active control system offers the advantage of inertial stabilization with the potential for high pointing accuracy. It is the best method for maneuvering which allows for high precision and adaptability to perform several different missions. A disadvantage of the system is that six possible control directions (pitch,roll, and yaw) must be maintained. Also a two-axis sun sensor is required due to the absence of rotation.

CONTROL MODES

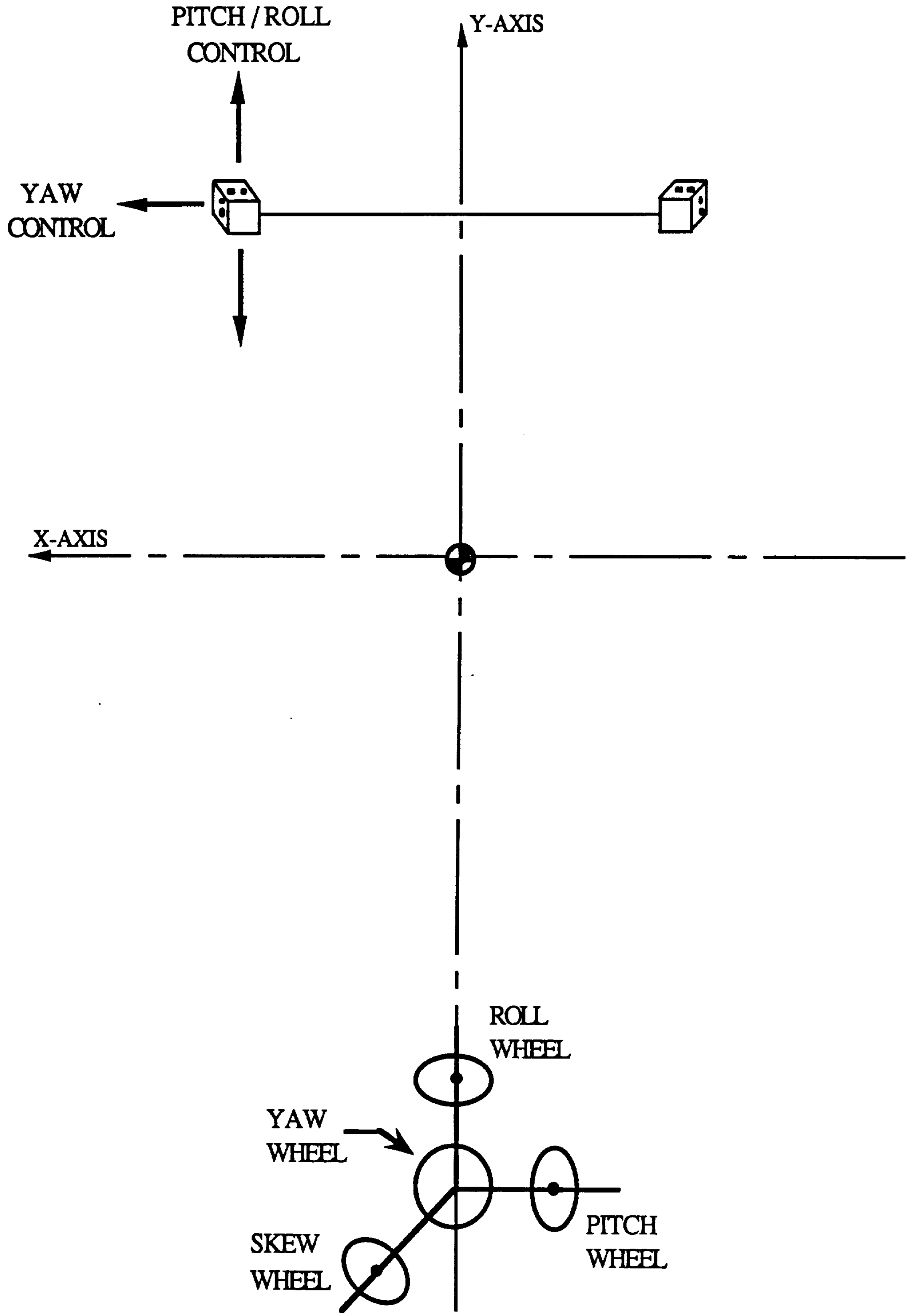
The control modes for the various phases of the mission are:

1. Attitude acquisition mode
2. Cruise mode
3. Trim maneuver mode
4. Orbit insertion mode
5. Large maneuver mode

The first three modes rely primarily on sensor information and low maneuvering thrust, while the last two Require both sensor information and considerable auxiliary propulsion. Further analysis of the control modes is discussed in terms of AACS hardware selection and performance in the following section.

AACS HARDWARE

To fulfill the requirements of long mission life, pointing accuracy, and total redundancy a dual control actuation system was selected. The system includes twelve .005 newton thrust mercury ion thrusters (4 on each axis with 6 in operation and 6 redundant), and a reaction wheel assembly. Figure 2 shows an operating schematic of the system. During the initial attitude acquisition phase of the mission both systems will be


PITCH / ROLL
CONTROL
Y-AXIS
YAW
CONTROL
X-AXIS
ROLL
WHEEL
YAW
WHEEL
PITCH
WHEEL
SKEW
WHEEL

used to increase the spacecraft maneuverability. Throughout the 12-14 year cruise phase The reaction wheel assembly will provide primary control actuation, with the thrusters used for momentum desaturization and trim maneuvers. The final stage of the mission requires fine pointing of the science instrumentation and the antenna, which is control by the more stable reaction wheel assembly. Again the thrusters could extend maneuverability or take over primary actuation if necessary. This configuration satisfies the reliability requirement through total redundancy, and minimizes the auxiliary propulsion fuel usage during the cruise phase while maximizing maneuvering capability throughout the mission.

The attitude acquisition system includes a pair of two-axis sun sensors mounted on either side of the payload platform, which provides a 4Π steradian view. A celestial sensor assembly utilizing six detector slits in a spoke configuration is mounted at the far end of the payload platform to allow an unobstructed field of view for continuous star reference. Also, an inertial measurement unit containing three rate integrating gyros (2 for three redundancy) is located in the AACS cylinder centered along the y-axis of the spacecraft, which provides displacement information through rate integration to the control computer.[2] Figure 4 shows the location of the attitude acquisition system on the payload platform and table 2 describes the AACS components and gives the total AACS mass.

All selected hardware has been space tested, particularly in the Defense Meteorological Satellite Program Which satisfies the "off the shelf" requirement.[2] Also the system is capable of switching attitude acquisition responsibilities to different sensor configurations in the event of a component malfunction, which provides for total system redundancy.

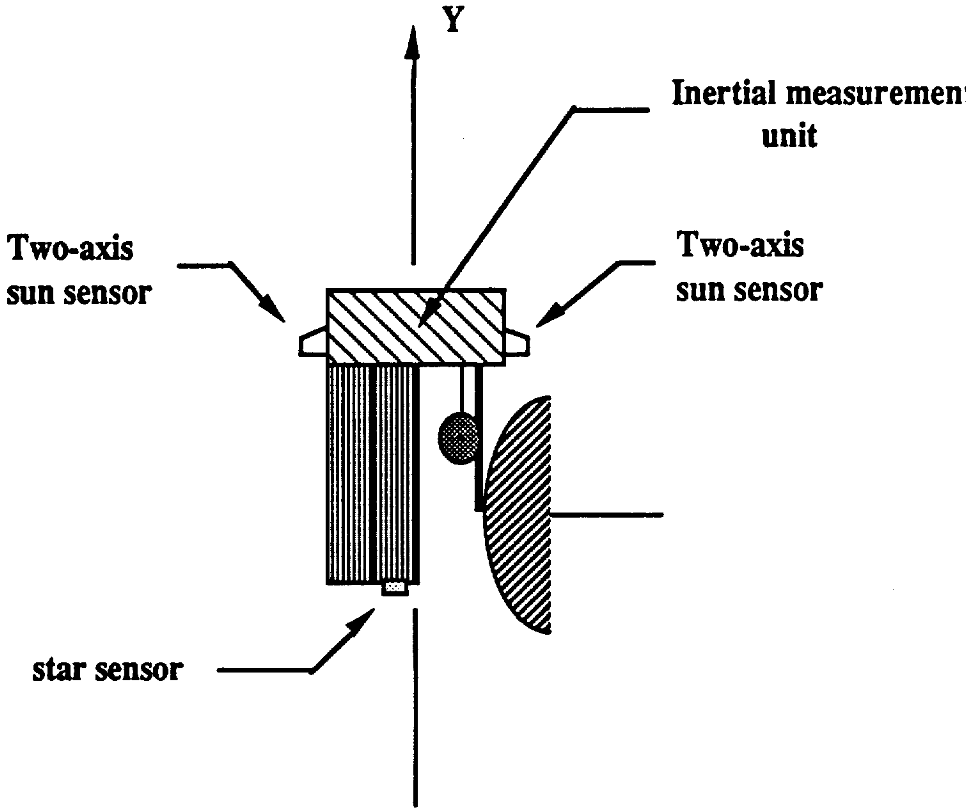


Figure 4
Sensor placement

AACS HARDWARE DESCRIPTION

Table 2

1. Inertial measurement unit: (Honeywell)	56.0 W	15.0 Kg
2. Sun Sensor (2) (SAGE,HCMM)	3.0 W	1.6 Kg
3. Star Sensor Assembly (Honeywell)	1.5 W	4.3 Kg
4. Reaction Wheel Assembly (4) (RCA AED)	16.0 W	25.6 Kg
5. Mercury Ion Thrusters (12)	----Included in propulsion subsystem-----	
TOTALS	76.5 W	46.5 Kg

SYSTEM INTEGRATION

A primary requirement of the AACS is integration with the other subsystems. The science and communication subsystems both rely heavily on the the AACS for antenna and instrument pointing. Antenna pointing accuracy must be in the range of .5 to 10.0 degrees, while instrument pointing requires an accuracy range of .35 to 2.0 degrees. The three-axis stabilized design meets the requirements with a pointing capability of .001 to 1.0 degrees depending on selection of and condition of the sensors. Sun shielding is another important concern of the science subsystem during the early phase of the mission. The initial solution to this problem was to orient the spacecraft such that the antenna would shield the instruments, but this approach was rejected in favor of of enclosing the sun sensitive instruments in a hinged shield box when not in use. Finally the configuration must be such that the center of mass does not change as fuel is expended. To avoid this problem the spherical fuel tank is located directly on the y-component of the vehicle center of mass.

DESIGN PROBLEMS

External and internal torques on the spacecraft can cause undesirable structural stresses and changes in attitude if not counteracted. The three-axis active control system is particularly sensitive to environmental disturbances such as meteoroid bombardment and solar radiation. Also impingement forces from the ion plume effects and internal torques due to actuator operation tend to take the spacecraft out of a stable configuration. The spacecraft will oppose these disturbance forces with occasional trim maneuvers to return the vehicle to the desired orientation.

Another problem imposed by the long mission life is gyro drift. To correct this deviation the star sensor is used to obtain an exact position from the last best position estimate from the gyro. This correction is returned to the gyro and actuation is implemented if necessary.[4] Other problems encountered include the required life of the AACS components, which is satisfied by total system redundancy, and mercury contamination of the sensor surfaces from the main thrusters, which is minimized as the distance between these areas increases. The more sensitive instruments require shielding which is accomplished with the enclosed science box and small shields above (towards the propulsion section) the star and sun sensors.

DISCUSSION

The final design selection meets all of the specified and implied AACS requirements, and should provide an excellent attitude acquisition and maneuvering system for a mission of this type. The mission is limited only by the lifetime of the system hardware, which should increase in the future. The AACS is particularly effective for spacecraft maneuverability which is necessary to fulfill several different missions. Future research should focus on improved autonomous control capability, the radiation effects on C^3 and science systems, and long life reactors capable of powering a spacecraft for ten or twenty plus years.

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PROPULSION AND POWER

A.) REQUIREMENTS

B.) METHOD OF ATTACK

C.) SYSTEM

1.) PRIMARY THRUSTERS
- a.) POSSIBLE THRUSTERS
- b.) THRUSTER SELECTED
- c.) PROPELLANT
- d.) ION DYNAMICS
- e.) SPECIFICATIONS

2.) ATTITUDE AND ARTICULATION THRUSTERS

3.) POWER SOURCE

4.) INTERACTIONS WITH OTHER SUBSYSTEMS

D.) PROBLEMS

E.) EQUATIONS

F.) REFERENCES

A.) REQUIREMENTS

1.) Should use off the shelf hardware, nothing which has not been developed by 1999.

2.) Must be ready before 2010.

3.) Should optimize performance, weight, and cost.

4.) Should be reliable and easy to operate.

5.) Must be able to withstand any environment it may encounter.

6.) Must have a design lifetime to carry out its mission plus a reasonable safety factor.

7.) Nothing in the design should preclude it from performing several possible missions.

8.) Design will stress simplicity, reliability, and low cost.

9.) Exceptions to proposed technical requirements should be identified and justified.

10.) Primary thrusters must be able to deliver to Plutoian Orbit.

12.) Propulsion and Power subsystems must not interfere with other subsystems.

13.) Power subsystem must be able to deliver the power required by all other subsystems at any given moment.

B.) METHOD OF ATTACK

The general process that I followed when I was designing the various components of the propulsion and power subsystem is what I call my method of attack. The first thing was to develop a fundamental understanding of various types of ossibilities for a given component. Next, I evaluated the pros and cons of each candidate for that component, and how they related to the needs and objectives of the mission. By process of elimination, I then determined which candidates may be realizable. Next, I investigated the realizable candidates in depth, and determined which one is most suitable for the given mission. Finally, I continued to develop, and address problems related to the candidate decided upon until the final design is complete.

C.) SYSTEM

1.) PRIMARY THRUSTERS

In determining the type of primary thrusters, several factors were considered. First, the system should make efficient use of its propellant. The common measure of propellant efficiency is specific impulse(Isp) which is defined as the ratio of thrust to mass flow rate of propellant. Thrusters with high values of Isp have high exhaust velocities which translates to a high amount of energy in there exhaust streams. This allows such thrusters to move a more massive payload with less propellant. The second factor is thrust. Systems with higher values of thrust will be able to make journeys in less time for a given type of trajectory, either low thrust or impulsive. In addition, systems with high enough thrust to use impulsive velocity change trajectories have the benefit that their trajectories are computationally much simpler than low thrust trajectories. The third factor is the ease and cost of producing the system. In expensive systems which have been or can easily be developed and tested are preferred. The forth factor is additional mass associated with the system. Though a system may use its propellant efficiently, the associated mass may make the system as a whole inefficient compared with other options.

a.) POSSIBLE THRUSTERS

The first type of thruster considered is the chemical rocket. Solid chemical rockets have high thrust, but low Isp. In addition they cannot be throttled. Certain liquid propellants have an adequately high Isp to be used as a primary thruster on a journey of this length. However, the mass of the payload would be limited. Both solid and liquid chemical rockets have the benefit that they have already been developed, and flight proven many times.

The second type of thruster is the electrically propelled rocket. This includes electrothermal, electrostatic, and electromagnetic thrusters. These types of thrusters are capable of attaining very high values of Isp, but generally have low values of thrust. One drawback to this type of propulsion is that it has not really been researched on an interplanetary scale. Another drawback is that electric methods of propulsion require large amounts of power. This power requirement has an associated mass which is large with respect to the rest of the system.

The third type of thruster is the nuclear rocket. Performance of nuclear rockets is limited by the fact that there is a limit on the maximum solid surface temperature that the reactor must operate within to ensure structural integrity. Thus, unlike the condition found in a chemical rocket where the energy release is within the propellant, the propellant temperature in nuclear rockets is restricted to being less than the wall temperatures, and hence less than that found within chemical rockets. Another drawback is that since the propellant passes directly through the core of the reactor, the exhaust stream is contaminated. Nuclear rockets also have additional associated mass penalties which come from the reactor.

The fourth type of thruster type is cold gas. This is simply the thermodynamic expansion of a cold gas. Cold gas thrusters have low values of Isp, but are reliable and have been flight proven many times.

Other types of thrusters are solar, and laser. Solar propulsion is ineffective at the great distances from the sun that will be characteristic of this mission. Laser thrusters, as of yet are not developed.(ref. 1,2,3,5)

b.) THRUSTER SELECTED

Upon evaluating the options, I decided to use an electrostatic thruster on the Phoenix probe. During 1980, Studies at the Jet Propulsion Lab focused on the application of nuclear electric propulsion(NEP) to outer planet missions. The study concluded that NEP was much better than other competitive technologies, and that a 100 kw(electric) system significantly out performed chemical propulsion systems for outer planet exploration.(ref. 2)

Since NEP has not been developed, In reality, It would be the case that many additional dollars would have to be spent on research, development and testing for this mission. This would make it very unappealing as the best best method for the mission. However, as stated in class by teaching assistant Andy Koepke, for this project, it may be assumed that the technology has already been developed, and that costs affiliated with research and development may be neglected.

The additional mass associated with the power system needed makes the benefits of this type of propulsion system unclear when the payload mass is small compared to the power system mass. In fact it is possible that the propellant mass for the Phoenix probe may even be higher than that for analogous chemically propelled missions. However the the real benefits of NEP comes from the fact that once the mass of the power subsystem is fixed, the marginal or additional amount of propellant required for a given marginal payload mass will be much less than that for a chemically propelled system. Since the RFP states that the system should be capable of performing several types of missions, it is very important that the system should have a capacity for a marginal payload. Also with the capability of taking greater payload masses to a destination also comes the capacity for designing better science experiments which would not be realizable with chemical propulsion. Though at its present status the phoenix mission may not appear to be the best choice in terms of money, it has the capacity of having added to it some very advanced science experiments, including possibly a lander, before its launch date. In addition, information gained on NEP from this mission will be very beneficial to future high energy deep space missions where propellant efficiency is crucial.

c.) PROPELLANT

Determination of propellant is based on several factors. First, the propellant should have a high nuclear mass, and a low ionization potential. This is because the beam thrust is proportional to the square of the mass to charge ratio. Second, the propellant should be easily stored. This is especially important on missions of comparable duration to that of the Phoenix mission. Third, the propellant should be environmentally safe, non corrosive, and have minimal effects on other subsystems. Fourth, the propellant should yield a high thruster efficiency. (ref. 1,2,5)

One possible propellant is cesium. Cesium has a high mass to charge ratio, but is highly corrosive. Thus, it would be hazardous to both the environment as well as the other subsystems. Another possible propellant is xenon. Xenon is environmentally safe, and easily stored. However, it is expensive and rare. In fact there may not be enough currently available to make this one trip. Though xenon is a prime candidate for earth orbital transfers, there is simply not enough to make it practical for missions comparable in length to the Phoenix Mission. Another inert gas which could be used is argon. Argon is also environmentally safe, but is difficult to store. In addition, argon is more abundant than xenon. The final propellant considered was mercury. Mercury yields the highest thruster efficiency of those propellants considered. In addition, it is easily stored. The main problem with mercury is that it is poisonous. Since only a small fraction of the mission will be spent near the earth's atmosphere, environmental contamination is not a big problem.(ref. 2) This coupled with the fact that it best satisfies the guidelines used to evaluate the various propellants, makes mercury the propellant selected.

The sizing of the propellant tank was done by starting with the assumed value for the total mass of the mercury required which is about 12,000 kg. Next, the density of mercury was obtained, and turned out to be 13,800 kg/m^3. The volume required to contain the mercury was then computed by dividing mass by density. This gave a propellant volume of 0.87 cubic meters. Since a sphere is structurally more sound than a cube, the propellant will be contained in spherical tank of radius 0.592 meters.

c.) ION DYNAMICS

The method which will be used to generate ions will be electron bombardment. The neutral mercury or plasma, will be passed through a cylindrical anode. Surrounding the cylindrical anode will be a solenoidal coil which will be used to generate an induced magnetic field in the direction of the plasma flow. At the center of the cylindrical anode will be a heated filament cathode which will be the source of electrons. The filament will be heated by passing an electrical current through it. As a result, the heated filament will bleed of electrons. The free electrons will be accelerated radially outward by the cylindrical anode. The presence of the magnetic field will give a tangential force acting on the electrons making them spiral outward toward the anode, increasing the likelihood of them hitting a mercury atom before they reach the anode. The collision between the electron and the neutral mercury atom will produce the ion.

Once the ions are produced, they will then be subjected to an electrostatic potential difference. They will be accelerated toward an electrode which is at a lower potential. When the ions reach the accelerating electrode, they will be at their minimum potential, and have their maximum kinetic energy. As their momentum carries them past the electrode, they will be accelerated back towards that electrode, and will begin to lose their kinetic energy. Therefore it is necessary to recombine the ion stream with an electron stream in order for the ions to retain their momentum. Ideally one would want to recombine the ion stream with electrons at the point of lowest potential. However, trying to do so will result in the electrons diffusing into the acceleration field. Thus, there is an optimal distance from the electrode that the electron stream should be recombined with the ion stream. I, however am unable to compute this optimal distance. The electron stream used to neutralize the ion stream will be produced by the same method as the one in the ion source, using a heated cathode filament.(ref. 1)

e.) SPECIFICATIONS

Since a thruster comparable to those which will be used on the Phoenix probe has never been built, it is difficult to say how one would perform. most of these results were obtained from tables, or from crude approximations from similar data calculated by the Jet Propulsion Laboratory. The information has been combined from several sources, and in some instances represents the state of the art system which may not be attainable.

(ref. 2)

AVERAGE THRUST..........0.5 NEWTONS
SYSTEM THRUST..........2.0 NEWTONS
SPECIFIC IMPULSE..........5000 SECONDS

BEAM DIAMETER...30 CM

THRUSTER LIFETIME..125,000 HOURS

POWER REQUIRED/THRUSTER...20 KW(E)

NUMBER OF THRUSTERS...6

NUMBER OF OPERATIONAL THRUSTERS...............................4

MASS/THRUSTER...106 KG

DRY SYSTEM MASS..636 KG

PROPELLANT MASS...12000 KG

WET SYSTEM MASS...12636 KG

3.) ATTITUDE AND ARTICULATION THRUSTERS

The thrusters which will be used for controlling the attitude and articulation of the spacecraft, like the primary thrusters, will be Ion rockets. They will be very similar to the primary thrusters conceptually, but will be on a smaller scale. In order to control the attitude of the spacecraft, six thrust vectors will be needed. For each direction two thrusters will be present. This makes a total of 12 AA thrusters, 6 operational, and 6 for redundancy.

(ref. 3)

AVERAGE THRUST..0.005 NEWTONS

SPECIFIC IMPULSE...2650 SECONDS

BEAM DIAMETER..8 CM

THRUSTER LIFETIME..>15000 HOURS

POWER REQUIRED/THRUSTER..0.2 KW(E)

NUMBER OF THRUSTERS...12

MASS/THRUSTER..28 KG

SYSTEM MASS...340 KG

4.) POWER SOURCE

It is clear from the specifications for the ion rocket that a great deal of electrical power will be required. Specifically, to run the four thrusters will require 80 kwe. In addition, power must be reserved for other subsystems onboard Phoenix. Development of such a power source has been pursued intensely in recent years. The main product of this research and development is the sp-100 nuclear reactor. The sp-100 has an electrical power output of 100 kw. This will fulfill the 80 kw required by the four operational thrusters, and leave 20 kw for other subsystems. The other subsystems should not require nearly that much power. The reactor lifetime is about 7 years at maximum power output, and longer for output less than maximum. Since the sp-100 onboard the phoenix spacecraft will be operating at about 82%, it will be assumed that the reactor lifetime is 10 years. Since the mission is expected to take about 15 years, it will be necessary to bring two reactors. Another benefit of using NEP is that it allows the other subsystems as much as 100 kw for several years after arrival at the destination. Thus science projects requiring large amounts of power can be conducted over long periods of time.

(ref. 4)

THERMAL POWER OUTPUT	1.4 MW(T)
ELECTRICAL POWER OUTPUT	100 KW(E)
REACTOR LIFE AT MAXIMUM OUTPUT	7 YEARS
REACTOR LIFE AT 82 %	10 YEARS
REACTOR MASS	640 KG
SHIELD MASS	860 KG
HEAT TRANSPORT MASS	445 KG
REACTOR I & C MASS	210 KG
POWER CONVERSION MASS	315 KG
HEAT REJECTION MASS	835 KG
POWER CC&D MASS	370 KG
STRUCTURE MASS	265 KG
SYSTEM MASS	4600 KG

5.) INTERACTIONS WITH OTHER SUBSYSTEMS

In addition to the thermal and plume interactions which are associated with chemical propulsion spacecraft, there are also reactor neutron and gamma fluxes as well as electromagnetic fields associated with an electric propulsion spacecraft. Thermal interactions are minimized by the fact that the spacecraft subsystems are integrated along a thermal gradient. The high temperature reactor at one end, intermediate temperature equipment in the middle, and low temperature science instrumentation at the other end. Other interactions, as well as thermal, are reduced by putting distance between the interactive elements.(ref. 2) Since I do not really have an understanding of most of these interactions, details on the configurations required by two interactive elements was obtained from examples done by the Jet Propulsion Laboratory.

D.) PROBLEMS

Many problems have come up during the design of Phoenix and its propulsion system. One problem is the political pressure of having a nuclear reactor onboard a space vehicle. It will be difficult to convince the public that the reactor will remain safe in the event of an accident at launch even though it has been verified to remain safe in almost any type of disaster. Another problem has been demonstrating the true effectiveness of NEP. Almost everything in the design of a space mission is geared to the optimal level of Chemically propelled rockets. When NEP performs at this level, it appears to be an inferior method of propulsion. Thus, in order to sell the Phoenix program it may be necessary to turn it up a notch in mission objectives as to utilize the full potential of NEP. I have encountered many problems in the design of the Phoenix propulsion system. Some of these problems are that details related to this type of propulsion are difficult to find if they even exist, and often data conflicts depending on the source. Another design problem is that optimizing computation dealing with many aspects of the design are difficult, or at least exceed my level of education. Thus, I am often required to go on blind faith as to the validity of some of the results.

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SUMMARY OF ABBREVIATIONS

C^3	COMMAND, CONTROL, AND COMMUNICATION
AI	ARTIFICIAL INTELLIGENCE
CDS	COMMAND DATA SYSTEM
R.F.P.	REQUEST FOR PROPOSAL
D.O.D.	DEPARTMENT OF DEFENSE
RH32	RADIATION HARDENED 32-BIT PROCESSOR
GVSC	GENERAL PURPOSE VERY HIGH SPEED INTEGRATED CIRCUIT SPACEBORNE COMPUTER
HGA	HIGH-GAIN ANTENNA
LGA	LOW-GAIN ANTENNA
HPBW	HALF-POWER BEAMWIDTH
RTI	REAL-TIME INTERRUPT
DSN	DEEP SPACE NETWORK
N.E.P.	NUCLEAR-ELECTRIC PROPULSION

Command, Control, and Communications

Group 7

The document "Request for Proposal for an Unmanned probe to Pluto" lists requirements which must be understood and complied with if this preliminary design work is to be useful in the ongoing design process which will result in the eventual construction of an unmanned probe to be sent to the celestial body known as Pluto. While all requirements listed in the R.F.P. (Request for Proposal) pertain to the development of the C^3 (Command, Control, and Communication) subsystem, only those requirements which most directly apply to the C^3 subsystem are explicitly discussed in this portion of this document. A table listing requirements that are of particular importance is shown below (table $C^3$1).

TABLE $C^3$1 : REAL AND IMPLIED REQUIREMENTS

- -Select microprocessors and peripherals for Phoenix
  - -Select software to optimize spacecraft autonomy
- -Select and size communications hardware for mission that allows transmission at adequate speed with high quality
- -Develop overall communications plan, including ground communications
- -Recognize and defend against pointing problems and communications loss
- -Optimize mass, size, strength, reliability, cost, and performance
- -Components must be space qualified
- -Provide sufficient computer speed and storage to implement Artificial Intelligence
- -Provide sufficient data storage for scientific objectives
- -Utilize components available no later than 1999
- -Design hardware to be redundant when possible
- -Design software to be as robust and autonomous as possible
- -Transmit and receive command, telemetry, tracking and science data

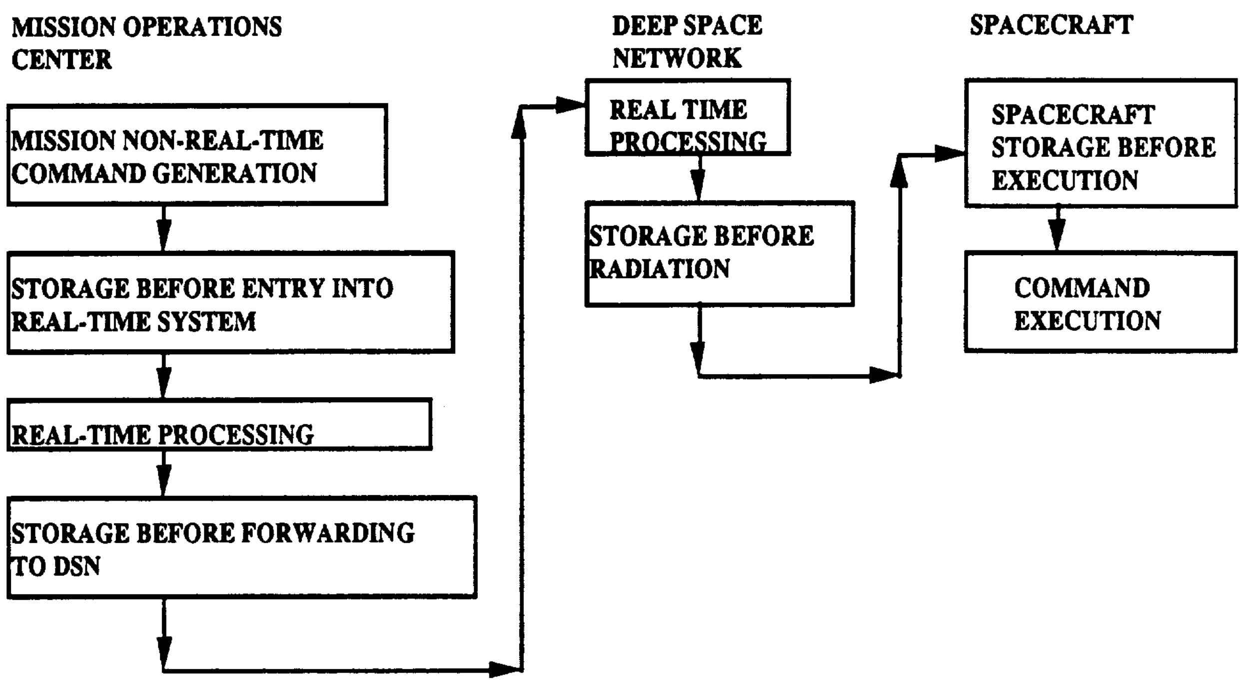
To comply with the requirements in the R.F.P. a modified design-by-design approach was followed. Reference materials pertaining to the C^3 subsystem were found without excluding references that did not specifically pertain to the exact R.F.P. requirements. These references were used to gain a general knowledge of the C^3 subsystem on past and proposed space missions. The general knowledge from these sources was then used to interpret the design requirements that were imposed by the R.F.P. and by the evolving designs of the other Phoenix subsystems. This synthesis of general knowledge, R.F.P. requirements, evolving Phoenix probe design, and information attained from AAE 241 class notes shaped further research and design work as it applied to the C^3 subsystem. After an initial design was reached, the subsystems were consciously integrated and an iterative process was begun to optimize the overall performance of the Phoenix .

A major responsibility of the C^3 subsystem design team is to select computer equipment to be used on the Phoenix. Driving factors in the selection of the computer equipment for the Phoenix probe were dominated by the desire for greater autonomy than previously attempted in spacecraft design. This desire for autonomy, specifically through the implementation of AI (Artificial Intelligence), requires that the computer system for the Phoenix must be faster and have more memory than past NASA interplanetary probes. Therefore, it is important that the fastest microprocessors available be selected and combined with a large amount of internal memory and external storage. Three microprocessors were seriously considered for use in the development of the Phoenix computer system. They include the D.O.D. (Department of Defense) developed RH32 (Radiation Hardened 32-bit Processor), the Department of Energy's Sandia

Application 3300, and the D.O.D. developed GVSC (General Purpose Very High Speed Integrated Circuit Spaceborne Computer). The RH32 was selected due to the high speed of its 32 bit architecture and the added reliability its radiation hardening will afford in the environment of our Nuclear-Electric Propulsion system and the environment of Venus or Jupiter in the event of a gravity assist fly-by. The entire computer system will be loosely based on the multiply redundant CDS (Command and Data subsystem) used on the recent Galileo space probe. Six RH32 microprocessors in combination with eight memory units have been selected to be linked by a bus running at approximately 400 KHz with a RTI (Real Time Interrupt) running at approximately 15 Hz (a configuration similar to what was used as a part of Galileo). The internal memory can be backed up to and loaded from an external storage system utilizing the space proven magnetic tape that NASA has used on numerous past interplanetary missions.

This computer hardware will be used to implement an artificially intelligent autonomous system that has been referred to as an "intelligent associate".[1] The capabilities of an AI system, which are expected to be available by the time of the Phoenix mission, will make the mission more productive and versatile than it could be without the use of AI technology. With an approximate round trip light time to Plutonian space in the neighborhood of eleven hours, the Phoenix must be able to carry out its mission without constant supervision from earth. The time that it takes for a signal to be sent to the Phoenix, demonstrates the correctness of the R.F.P. requirement that the spacecraft design should maximize autonomy and use AI wherever possible. Advantages gained by the implementation of autonomous systems in spacecraft design include a reduction of mission operation costs, an increase in overall mission productivity, and an increase in mission success probability. Continuing work in the field of AI will provide many possible capabilities with which the Phoenix could be equipped. Capabilities which will be useful and practical for implementation in the Phoenix Probes CDS include distributed control of multiple subsystems, fault prediction and analysis, automated real time planning and replanning, and a reasoning/learning supervision of on-board systems. Using sets of "heuristic algorithms" and priorities the Phoenix Probes on-board computer systems will independently react to the changing environments that the craft will encounter. Through an integration of science data, engineering data, tracking, telemetry, and its programming, the Phoenix probe will respond to threatening situations and unique opportunities for scientific observation. The reprogrammable nature of current spacecraft computer components will also allow mission designers at earth a great deal of flexibility after the Phoenix has been launched. The R.F.P. states that the design of the spacecraft should not preclude its use for other missions, and the ability to reprogram the Phoenix computers is an important way in which this requirement is met. Much as the Voyager mission planners were able to send "patches" to deal with Voyager performance anomalies, so to will the Phoenix and Phoenix mission planners be able to respond to changing mission circumstances and requirements. The inclusion of eight memory units (more than twice the memory of Galileo) allows much more flexible control of on-board systems during different phases of the mission. When the program for a certain mission operation is no longer needed it can be backed up to magnetic tape or discarded altogether leaving room for new programs to be implemented in system memory. In the event that multiple hardware failures should occur, defeating redundant design considerations, the situation could be handled through the use of programming "patches" which could account for the new spacecraft performance characteristics. The extreme length of the light time from the Phoenix to earth during most of this mission also suggests the use of a "store and forward" command system.[2] In a "store and forward" system large blocks of commands are sent as a single communication to be received and verified before the execution of commands is begun, as seen in fig. $C^3 1$.

FIG $C^3 1$: STORE AND FORWARD COMMUNICATION SYSTEM



It should be noted that the use of an autonomous system and the "store and forward" technique need not preclude the use of near-real-time commanding of the Phoenix probe. A large amount of memory also allows redundancy in the gathering of scientific data for transmission to the earth. Copies of images or science data can be saved in memory or backed up to magnetic tape until confirmation of the reception of the data can be beamed back from earth, preventing the loss of important data taken during "one chance" scientific observations. It may also be noted that the choice of N.E.P. and an orbiter mission will greatly reduce the number of these "one chance" observations. It is necessary that the C^3 subsystem interact closely with all other on-board systems. The programs implemented as part of the CDS must be able to coordinate the activities of the power and propulsion subsystem; the attitude, articulation, and control subsystem; the thermal control system, and the science instrumentation subsystem. It is the responsibility of the on-board computer to transmit its commands and commands from earth to each of the other spacecraft subsystems.

It is also the responsibility of the C^3 design team to select and or design the components that will be used to communicate between the spacecraft and the earth. To accomplish this different communication systems were considered, including laser and traditional multi-frequency radio communication. Though technology for laser communications is developing quickly, the desire to use off-the-shelf components when possible suggested that the use of S and X-band communications with the earth would be most cost effective. Often in Spacecraft communication system design antenna gain and power required for communications must be painstakingly evaluated to find the ideal balance between communications performance and spacecraft mass. On the Phoenix probe the abundant power provided by the XP-100 reactor and the overall large mass of the spacecraft imposed new parameters to be evaluated in the choice of spacecraft antenna. The most important factor driving the size of the Phoenix probe antenna is the transmission data rate that will be required to beam the science data gathered by Phoenix back to earth. Antenna's from past NASA missions were examined to see if they might meet the communication needs of the Phoenix spacecraft as they interacted with its larger power system. Pointing difficulties for different portions of the mission suggested that multiple antennas might be included for use during different phases of the trip to Plutonian space. Interaction with the structures subsystem dictated that launch volume of the main HGA (high-gain antenna) could be minimized by using a folding system similar to that used on the Galileo mission. A comparison of different antenna types with respect to gain and pointing factors (HPBW, Half-power beamwidth) was made.

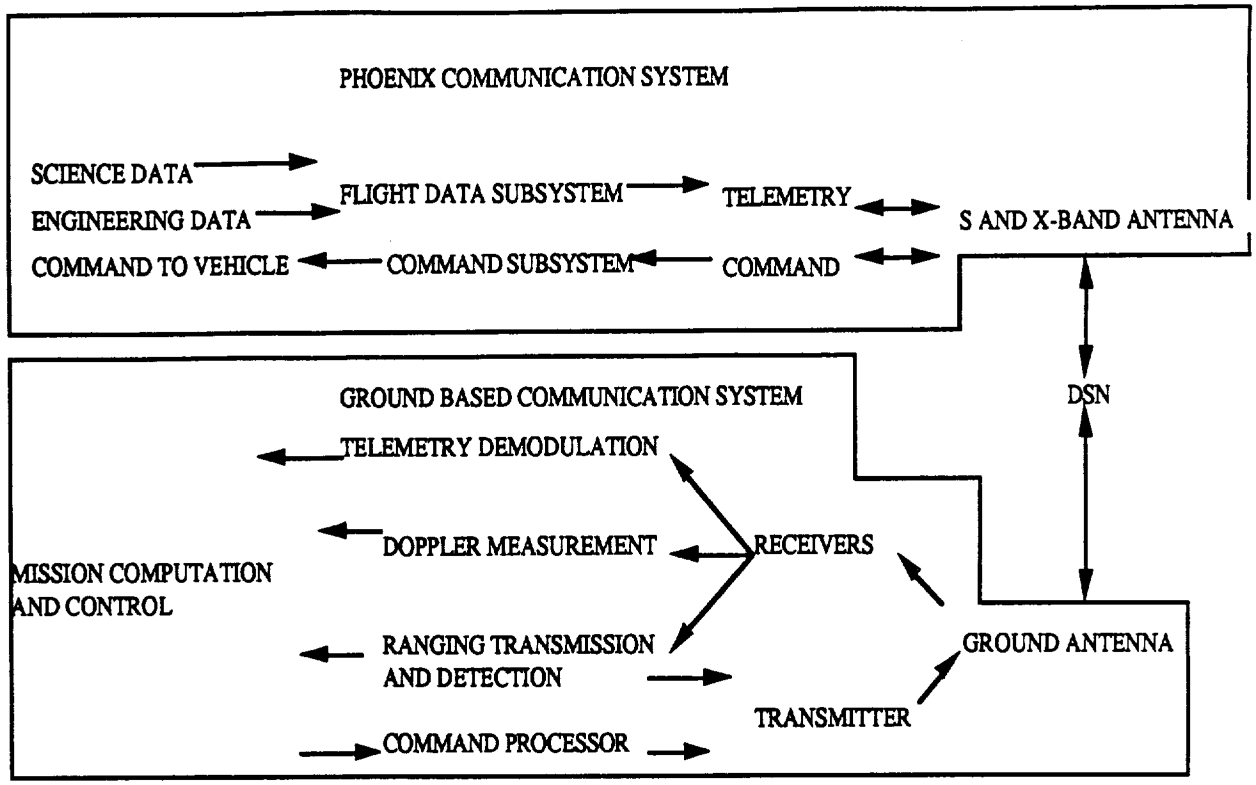
This information can be seen in table $C^3 2$. [3]

TABLE $C^3 2$: Antenna Type Comparison

Configuration	Gain above isotropic radiator	HPBW, deg
Isotropic radiator	1.0	360.0
Infinitesimal dipole or loop	1.5	89.9
Half-wave dipole	1.64	78.0
Paraboloid	6.3 to 8.8 (Area/wavelength*2)	60 to 70(wavelength// diameter)

The Galileo main parabolic HGA was chosen to be used as a part of the Phoenix with some minor redesign. It was estimated to be large enough to meet the data rate transmission requirements of the Phoenix probes science subsystem while still remaining small and light enough to be launched with the rest of the craft. The redesign would involve the use of lighter structural materials and antenna shielding, as the Phoenix HGA will not be used as a solar shield as it was on the Galileo mission. The Phoenix variant of the Galileo main antenna will fold to be stowed at launch as did its predicesor. The Phoenix HGA will communicate with the earth and DSN (Deep Space Network) using both X and S-band frequencies. The maximum power transmitted will be approximately 1 KW. This unprecedented amount of power is a result of the unusual nature of our nuclear power source. The deployed diameter of the antenna will be approximately 4.8 meters, so that a minimum amount of redesign will be required on the Galileo antenna while still fulfilling all the antenna requirements for the Phoenix probe. In addition to the parabolic HGA a smaller LGA (low-gain antenna) will be used as part of the Phoenix design. The 1 meter LGA will be a half-wave dipole antenna. The modest increase in antenna gain over an isotropic radiator is made up by the 78 degree pattern through which communication with earth can be maintained using the LGA. The ease with which the Phoenix probe could reattain contact with the earth in the event of some problem makes this secondary antenna an important tool for increasing the mission success probability. The LGA will also play an important role in the early phases of the mission when propulsion concerns may be more crucial than the pointing of instruments and the HGA. The large HPBW of the Phoenix LGA will allow the spacecraft to almost constantly transmit and receive engineering, tracking, telemetry, and command transmissions should they be necessary. Fig.$C^3$2 shows a representation of the Phoenix Communication subsystem. [4]

FIG $C^3$2 : PHOENIX COMMUNICATION SYSTEM



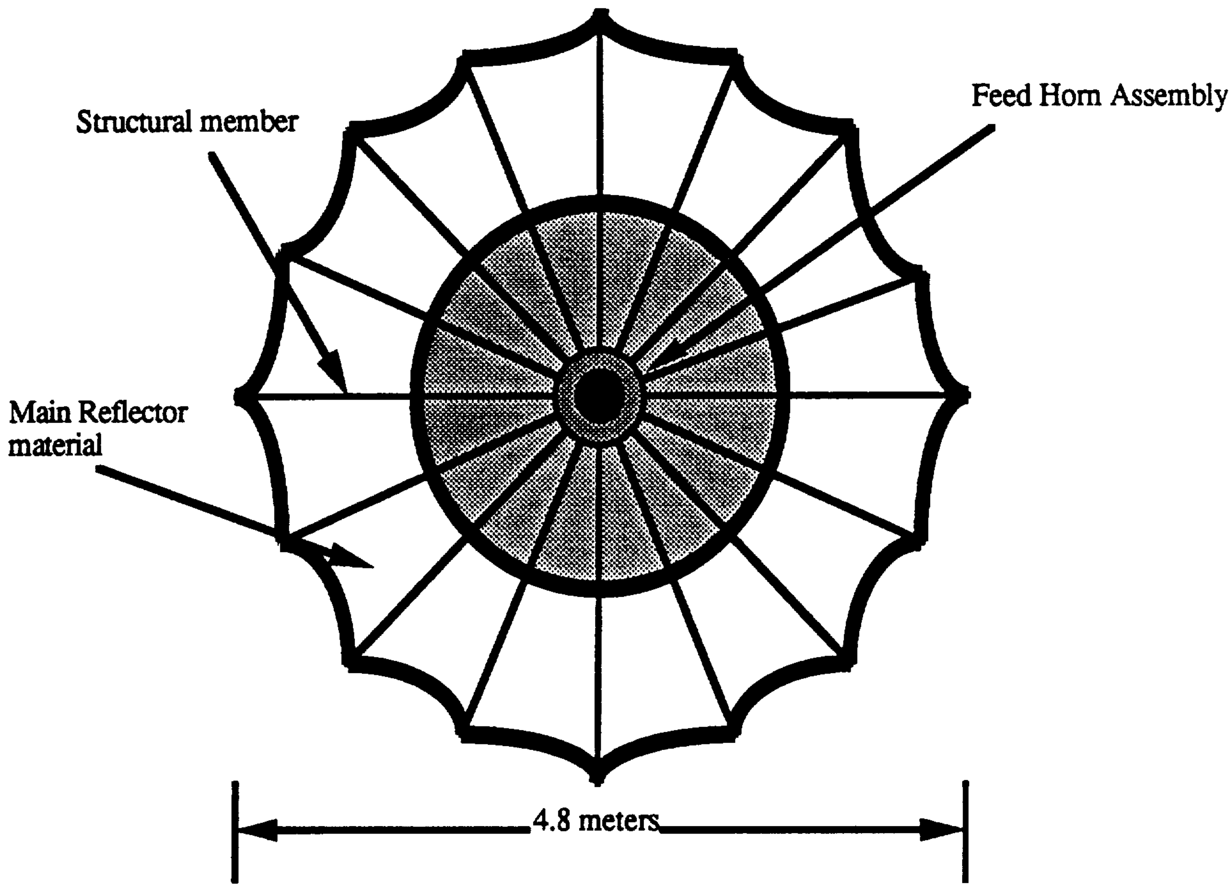
A unique and important consideration in the design of the Phoenix probe's communication system was the presence of the SP-100 nuclear reactor and mercury ion thrusters as part of the main propulsion unit. Though research into the effects of ion thrusters on a communication system of this type show that impact is slight (approximately a .2 K increase in antenna noise temperature)[5]., the general configuration of the Phoenix probe allows the communication system to be isolated from both the thrusters and the reactor by the main structural boom.

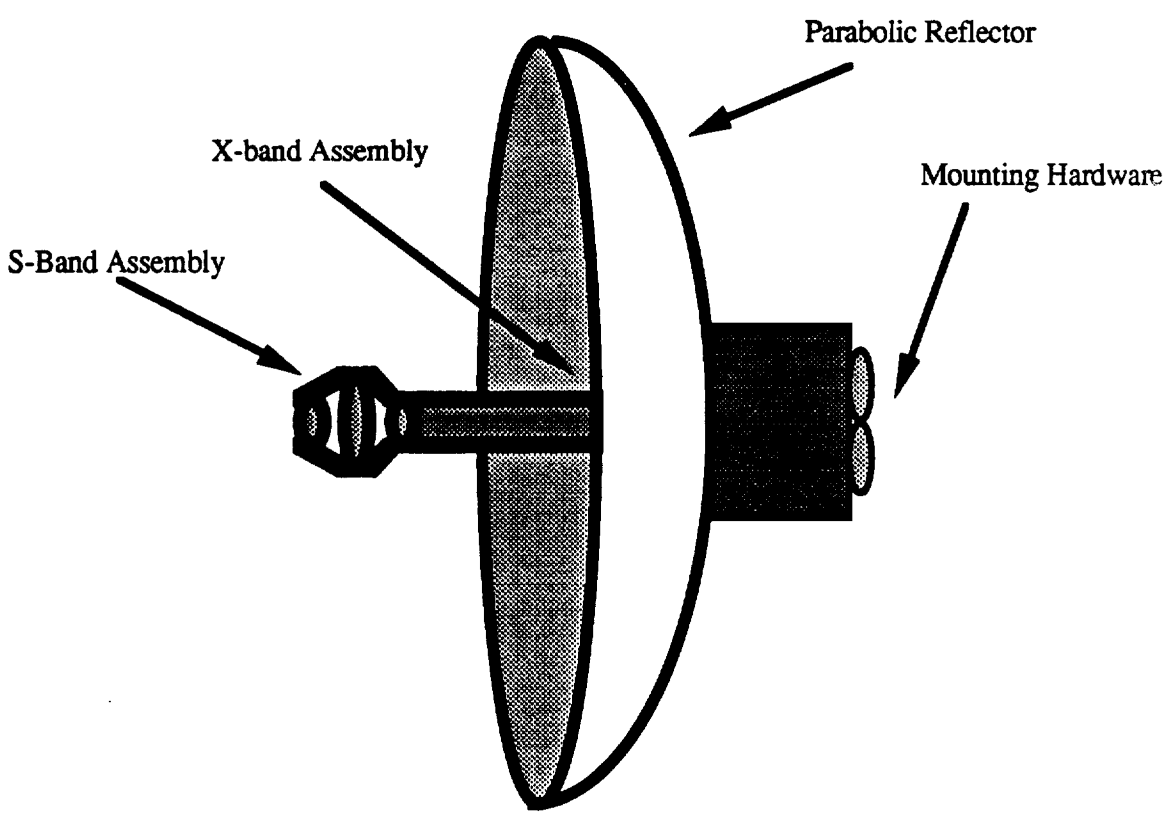
The design of the C^3 subsystem involved making many compromises between the performance of a given piece of equipment and other factors imposed by the R.F.P. and the interactions between the C^3 subsystem and others. The speed and storage capability of the computer system was maximized to allow for as complete as possible implementation of AI. The decisions regarding command procedures were driven by a need to make the Phoenix probe as autonomous as possible. Communication system choices were mainly dictated by the vast distances and amount of science data that Phoenix will beam to earth from its position orbiting Pluto. Major design problems that have been identified include the uncertainty about the conditions of Plutonian space, the interaction between the N.E.P. system and communications, the relatively long life required for this mission, and the great distance between the earth and Pluto.

The following page shows a graphic depicting the Phoenix HGA.

The next page shows a breakdown of the major component masses of the C^3 subsystem.

PHOENIX HGA


Structural member
Feed Horn Assembly
Main Reflector material
4.8 meters


Parabolic Reflector
X-band Assembly
Mounting Hardware
S-Band Assembly

COMMAND, CONTROL, AND COMMUNICATION MASS ESTIMATES

S/X BAND ANTENNA ASSEMBLY	5 KG
ANTENNA CABLING	4 KG
DATA STORAGE	19 KG
COMMAND DATA SUBSYSTEM	35 KG
MODULATION/DEMODULATION	10 KG
RADIO FREQUENCY SUBSYSTEM	50 KG
MAIN HGA	250 KG
HALF-WAVE DIPOLE LGA	50 KG

TOTAL MASS APPROX. 423 KG

References Cited

1. Technology for Future NASA Missions: Civil Space Technology
   AIAA/NASA Conference

2. Deep Space Telecommunications Systems Engineering 1983
   Joseph H. Yuen

3. Deep Space Telecommunications Systems Engineering 1983
   Joseph H. Yuen

4. Deep Space Telecommunications Systems Engineering 1983
   Joseph H. Yuen

5. Spacecraft Ion Beam Noise Effects
   G.L. Anenberg

APPENDIX A: EQUATIONS

POWER RECEIVED

$P_R = P_T + L_T + G_T + L_S + G_R + L_R$ IN DECIBELS

PARABOLIC ANTENNA GAIN

$G = 10\ LOG_{10}(\ .55\ (3.14\ DIAMETER/WAVELENGTH)^2$

SHANNON'S LAW

$B = W\ LOG_2(P_R/P_N + 1)$ = INFORMATION CAPACITY